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LOCKHEED MISSILES & SPACE COMPANY

A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION

SUNNYVALE, CALIFORNIA

FINAL REPORT
PAYLOAD EFFECTS ANALYSIS STUDY

30 June 1971

Contract NAS W-2156

LOCKHEED MISSILES & SPACE COMPANY

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FOREWORD

This Final Report describes the analyses performed and the results derived during the execution of the Payload Effects Analysis Study. This Study was performed by Lockheed Missiles & Space Company (LMSC), Sunnyvale, California, under Contract NAS W-2156, for the Office of Manned Space Flight, NASA Headquarters. The Study is part of a total economic analysis of the Space Transportation Systems being conducted for NASA by Mathematica, Aerospace Corporation and LMSC.

The LMSC study is concerned with determining the effects upon payload design, development and operations costs that could result from the use of future candidate launch vehicles, including the reusable Space Shuttle and Space Tug. Task 1 involved the selection of three representative satellite payloads. Task 2A consisted of a parametric cost optimization analysis and the estimation of target costs and design goals for low-cost payloads. During Task 2B, the baseline payloads were redesigned to take advantage of the cost savings by new transportation systems. Under Task 5, development plans and implementation costs were developed for the low-cost designs, and cost factors for reuse and refurbishment were provided to NASA. Under Task 3, the impact of system and subsystem standardization upon the cost of the composite mission model was investigated, and the feasibility of subsystem standardization was substantiated. Under Task 4, a payload designers' handbook, documenting the design approaches applied during the study, was prepared.

The study was conducted during the period of September 1970 through June 1971 under the supervision of Dr. R. M. Gray of LMSC and the direction of Mr. W. F. Moore of the Office of Manned Space Flight, NASA Headquarters, Washington, D. C.

LMSC and NASA gratefully acknowledge the assistance of the following NASA Centers and aerospace contractors for data provided for use during this study:

- Ames Research Center
- Goddard Space Flight Center
- Kennedy Space Center
- Langley Research Center
- Manned Spacecraft Center
- Marshall Space Flight Center
- Aerospace Corporation
- Mathematica, Inc.
- The Boeing Company
- Eastman-Kodak Company

English units were used in all calculations and estimates reported herein. Conversion to SI units are shown in parentheses or as noted conversion factors.

All costs shown herein are in constant 1970 dollars.

GLOSSARY OF TERMS USED IN THIS REPORT

Although generally accepted in a technical sense, many of the terms used throughout this report have various connotations within the aerospace community. Thus, as a guide to the reader, some of the basic terms are explained below.

PAYLOAD SYSTEM	is used to collectively describe the payload, the payload/launch vehicle adapter, and any separation devices required to effect a clean separation of the payload from the launch vehicle.
PAYLOAD	is a collective word used to describe the total operating entity, such as a satellite, that is launched into orbit by the launch vehicle; it comprises spacecraft and experiments but excludes launch vehicle related elements - such as adapters - that are non-functional in orbit.
BASELINE PAYLOAD	describes a representative example of a current unmanned payload used to provide a basis for the development of low-cost approaches and cost comparisons; those selected for the study were: OAO-B, SRS, Synchronous Equatorial Orbiter and Mars Orbiter. (The latter two were synthesized from the basic Lunar Orbiter).
LOW-COST PAYLOAD	refers to payload designs which were developed using low-cost approaches and techniques that are compatible with the cost-saving potential arising from use of the new launch vehicles.
LAUNCH VEHICLE	is the system (lower and upper stages) used to inject the payload into its specified low earth orbit and includes the exit fairing or shroud; specifically in the Payload Effects Study, three launch vehicle types are considered - current alternate expendable, new low cost expendable, and reusable Space Shuttle.
SUBSYSTEM	refers to the major functional elements of a payload, describing prime equipment categories; eight (8) subsystems are used to define the payload system: • Launch Vehicle Adapter/Interface • Experiments • Structures and Mechanisms • Electrical Power

- Stabilization and Control
- Attitude Control & Propulsion
- Communications, Data Processing & Instrumentation
- Environmental Control

MODULE — refers to a complete functional portion of a subsystem; a module comprises several components and interconnect electrical harnesses housed within a single structural box.

COMPONENT — an assembly such as a star tracker, transmitter, or similar. Components are assemblies of parts.

PART — a piece of hardware, a quantity of which are assembled into a single component; examples are: transistor, lens, shaft, etc. Parts categories considered are:

- High-reliability
- MIL-Spec
- Aircraft
- Commercial

PROGRAM TIME — defined for the purposes of this study to be 'the period of time over which a discrete set of observations or measurements are to be made or that a specific service is to be provided by a payload system'. This period of time would begin with the launch of the first payload through to the end of the requirement to perform that particular set of observations (or until requirements have been redefined so as to necessitate payload redesign). No restrictions are placed upon the number of payloads that may have to be launched in order to maintain the system and, in fact, it generally results that the optimum flight duration (from a cost standpoint) is less than the program time.

FLIGHT DURATION — is that period of time after launch (or on-orbit repair) until failure of a specific payload to perform its function and requires refurbishment or replacement; could also be referred to as 'Mean Time to Payload Failure'. In this study flight duration was one variable which was traded off with payload reliability and weight and cost to repair or refurbish in order to arrive at minimum program costs.

TOTAL PROGRAM COST — includes all costs accruing to the design, fabrication, operation, launch, and repair or maintenance of a payload (with the exception of internal government costs) in accomplishing the set of measurements required throughout the program time. In this study payload cost is broken down into non-recurring costs (RDT&E), unit recurring costs, and operations costs; in addition the total program costs include the (expendable) launch vehicle costs together with all its operations costs, or, in the case of the Space Shuttle, an apportioned share of the launch and mission costs and the costs of on-orbit repair or retrieval Space Shuttle flights.

NON-RECURRING COSTS (RDT&E)	accounts for those research and development and qualificatio costs associated with development to a point where the paylo enters production; it includes engineering design, developme and qualification testing and test hardware, GSE, production tooling, logistics, facilities and test articles.
UNIT RECURRING COSTS	represents the cost of material and equipment, hardware fabrication, assembly, sustaining engineering, acceptance testing, and necessary spares for production of a single flight system; no amortization of RDT&E costs is included.
FIRST UNIT COST	cost of production of the first unit.
AVERAGE UNIT COST	cost of production of an average unit. Learning curve may b applied if production rate and quantity justify.
OPERATIONS COSTS	are those recurring costs associated with operating the payload and include launch operations and mission operations directly concerned with the payload system itself, the cost of data retrieval and reduction, and sustaining engineering support to operation.
COST APPORTIONMENT	refers to the process of allocating the various cost categor totals to individual subsystems (or subsystem items) as deem relevant by thorough evaluation of the design; cost apportio ment is necessary only if the original cost data are not broken down to the required level.
NON-ALLOCATABLE COSTS	are those cost elements that cannot be genuinely apportioned to the subsystem level and, therefore, are not strictly trad able at the subsystem level. Examples are transportation fr factory to launch base, advanced planning and applications, remote site management, mockups, manufacturing planning and coordination, etc.

Section 1
INTRODUCTION

For the past several years, the National Aeronautics and Space Administration has been studying the economic merits of new space transportation systems. Considerable reduction in transportation costs have been projected, especially for reusable Space Shuttle concepts. It has been implied also that savings from payload design may significantly augment and even exceed the transportation cost reductions. NASA contracted Lockheed Missiles & Space Company (LMSC) to determine the impact of these new transportation systems upon the cost of design, development and operation of unmanned satellites under Contract NAS W-2156, the Payload Effects Analysis Study. This study was conducted by LMSC during the period of 21 September 1970 through 30 June 1971.

In order to evaluate the overall economic impact of low-cost space operations, NASA assembled a team structured to combine experience, analytical tools, and data banks of three contractors. The overall economic analysis was conducted by Mathematica, Inc., of Princeton, New Jersey, supported by Aerospace Corporation of El Segundo, California, contractor for the Integrated Operations Payloads/Fleet Analysis Study and by LMSC for the Payload Effects Analysis Study. Aerospace provided Mathematica with launch vehicle and payload performance and cost data for the estimated combined NASA and DOD traffic models with varying operational scenarios for the time frame of the implementation of the new transportation systems (1978-1990). LMSC, under the Payload Effects Analysis Study, provided Mathematica and Aerospace with detailed design, weight, reliability, cost and schedule data for three selected unmanned payloads representing a broad spectrum of size, cost and complexity. LMSC contributed to the study the background of over 10 years of work in both expendable and reusable launch systems and in development and successful operation of a large number of the nation's satellites. The interrelationship of the three studies is shown in Fig. 1-1.

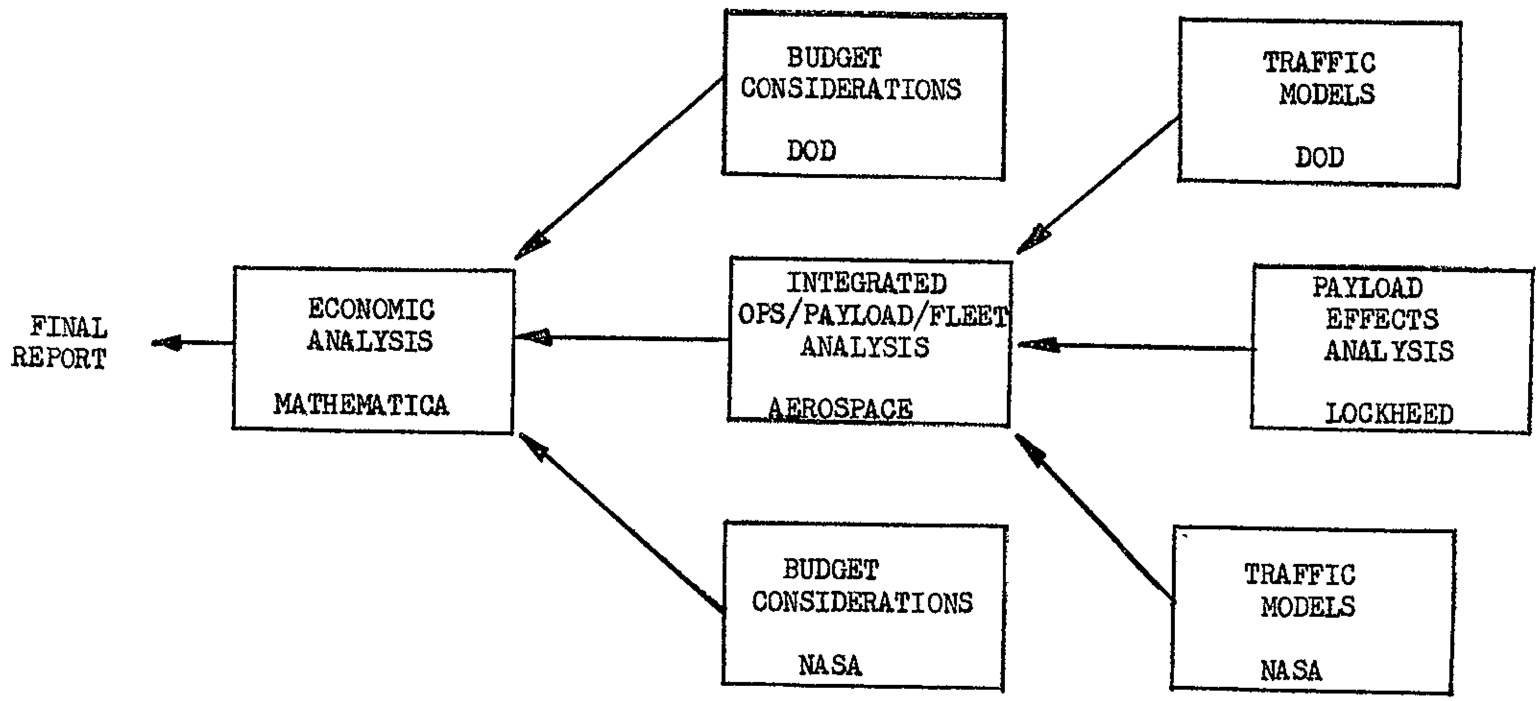

Fig. 1-1 Overall Economic Analysis Study Flow

The Aerospace and LMSC studies were under the technical supervision of Mr. William F. Moore, and the Mathematica Study was directed by Mr. Robert Lindley, both of NASA Headquarters, Office of Manned Space Flight. Messrs. Moore and Lindley were actively assisted in monitoring the combined study progress by a technical monitoring team comprised of key representatives of NASA Headquarters and the various NASA centers. The monitoring team further assisted the studies by providing data on advanced launch systems, payloads, mission and traffic models, and operations, and by providing critical technical review of study results. The members of the Technical Monitoring Team were:

Mr. William F. Moore
Special Assistant to Director
Space Shuttle Task Force
NASA Headquarters
COR Aerospace and LMSC Studies

Mr. Robert Lindley
Engineering and Operations Director
NASA Headquarters
COR Mathematica Study

Mr. Allen H. Sures
Office of Space Sciences & Applications
NASA Headquarters

Mr. Lawrence Hogarth
Advanced Plans Staff
NASA Goddard Space Flight Center

Dr. Michel Bader
Chief, Space Science Division
NASA Ames Research Center

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NASA Ames Research Center

Mr. Ernest Pritchard
Director, NASA Task A
Space Shuttle Economic Analysis
Aerospace Corporation

Mr. James O. Ballance
Mission & Payload Planning
Program Development
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Mr. William Huff
Space Shuttle Task Team
System Integration Office
NASA Marshall Space Flight Center

Mr. Harold Granger
Chief, Plans & Control Branch
NASA Manned Spacecraft Center

Mr. Jerry E. Hoisington
Space Shuttle Program
Payloads and Operations Office
NASA Manned Spacecraft Center

Section 2
SUMMARY

This document is the Final Report on the Payload* Effects Analysis Study conducted by Lockheed Missiles & Space Company (LMSC) for NASA Headquarters Office of Manned Space Flight under Contract NAS W-2156. This section of the report outlines the objectives of and approach to the study and provides a digest of the study analyses and results which are presented in later sections.

2.1 BASIC STUDY APPROACH AND SCOPE

The exploration of potential payload-related cost savings involves, by necessity, a departure from established ways of payload design, development, procurement, and operation. The study effort therefore was directed to both innovative and traditional cost-reduction methods.

Primary emphasis was placed upon the exploration of various cost-reduction measures on a set of selected payloads without altering the mission performance requirement. Toward the end of the study an evaluation was made of additional cost saving potential provided by sacrificing certain aspects of program peculiar payload design by mission or hardware standardization.

As the initial effort in the 9-month study, LMSC prepared a rather detailed Study Plan (LMSC-A973835) which described the study team organization, the task breakdown and schedules, and the technical approach. Highlights of these data are presented following.

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* Throughout this report, the word "payload" is used to designate the combination of the spacecraft and its experiments. See the glossary for other definitions.

2.1.1 Study Objectives and Groundrules

The basic study objective was to determine the contribution and effect of payload costs to the future NASA unmanned space programs of the 1978 to 1990 period. In executing this objective the following sub-objectives were established:

- Define design characteristics, method of operation, and costs for typical "low-cost" NASA unmanned payloads for use with new launch systems (expendable boosters and Space Shuttle).

- Derive differences in payload costs that can be anticipated as a result of introducing the new launch systems

- Identify minimum - cost payload approaches.

To obtain consistency in design and costing and to cover a reasonable scope of design variants, the following groundrules were established:

- Payload Performance - The performance of the newly-designed low-cost payloads was to be equal to the historical or "baseline" payload.

- State-of-the-Art Technology - 1970 technology was to be applied to payload hardware (the baseline payloads utilized hardware of the 1960's).

- Baseline Cost Data - All cost data was to be converted to 1970 dollars.

- Variants for Different Launch Vehicles - The low-cost payload designs were to be developed for each of three launch vehicles: (1) Alternate Current, (2) Low-Cost Expendable, and (3) Space Shuttle.

- Weight and Volume - The low-cost payload design was to assume essentially no weight nor volume constraints (except those imposed by the selected launch vehicles).

2.1.2 Study Organization and Task Breakdown

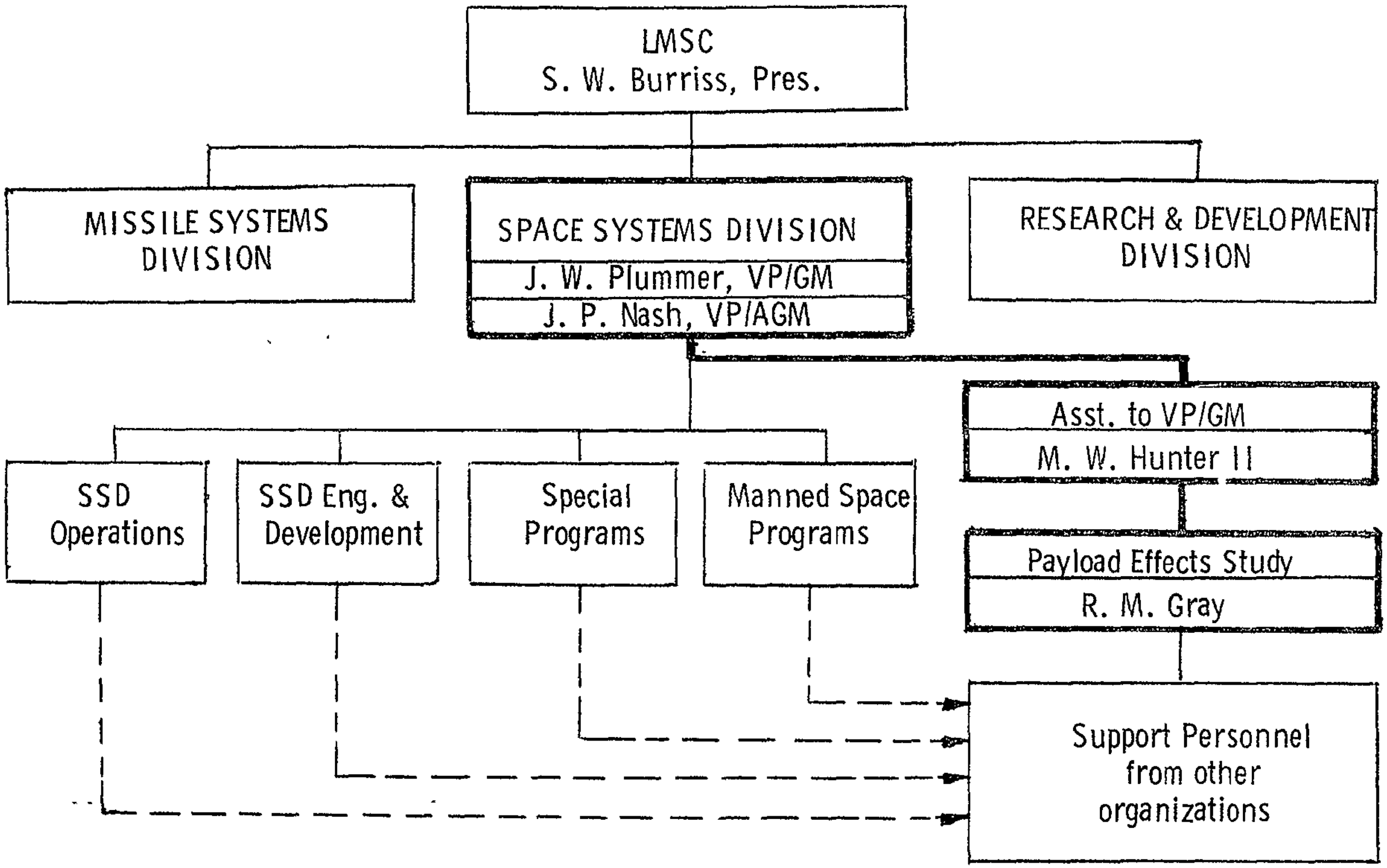
A nucleus team was assembled for the study effort from the Space Systems Division of LMSC and comprised senior engineers from both the spacecraft design areas (Special Programs and Engineering & Development) and the Space Shuttle design area (Manned Space Programs). Other supporting personnel were obtained from Planning, Cost Estimating, Manufacturing, and Product Assurance organizations as required. To obtain the required emphasis and managerial attention to the study and its results, a direct line of organization was established with the vice-president and general manager of the Space Systems Division. Figure 2-1 illustrates the study organization within LMSC.

The tasks of the study were set-up as shown on Fig. 2-2. These were described in detail in the aforementioned LMSC Study Plan. With the exception of Task 4, which is the preparation of the Payload Designers' Handbook, the analyses and results of each task is described in this Final Report. A separate document, LMSC-A990558 dated 30 June 1971, "DESIGN HANDBOOK FOR LOW-COST SPACE SHUTTLE PAYLOADS", has been prepared and is being submitted separately as a contract end-item document.

2.1.3 Payloads and Launch Vehicles Selected for the Study

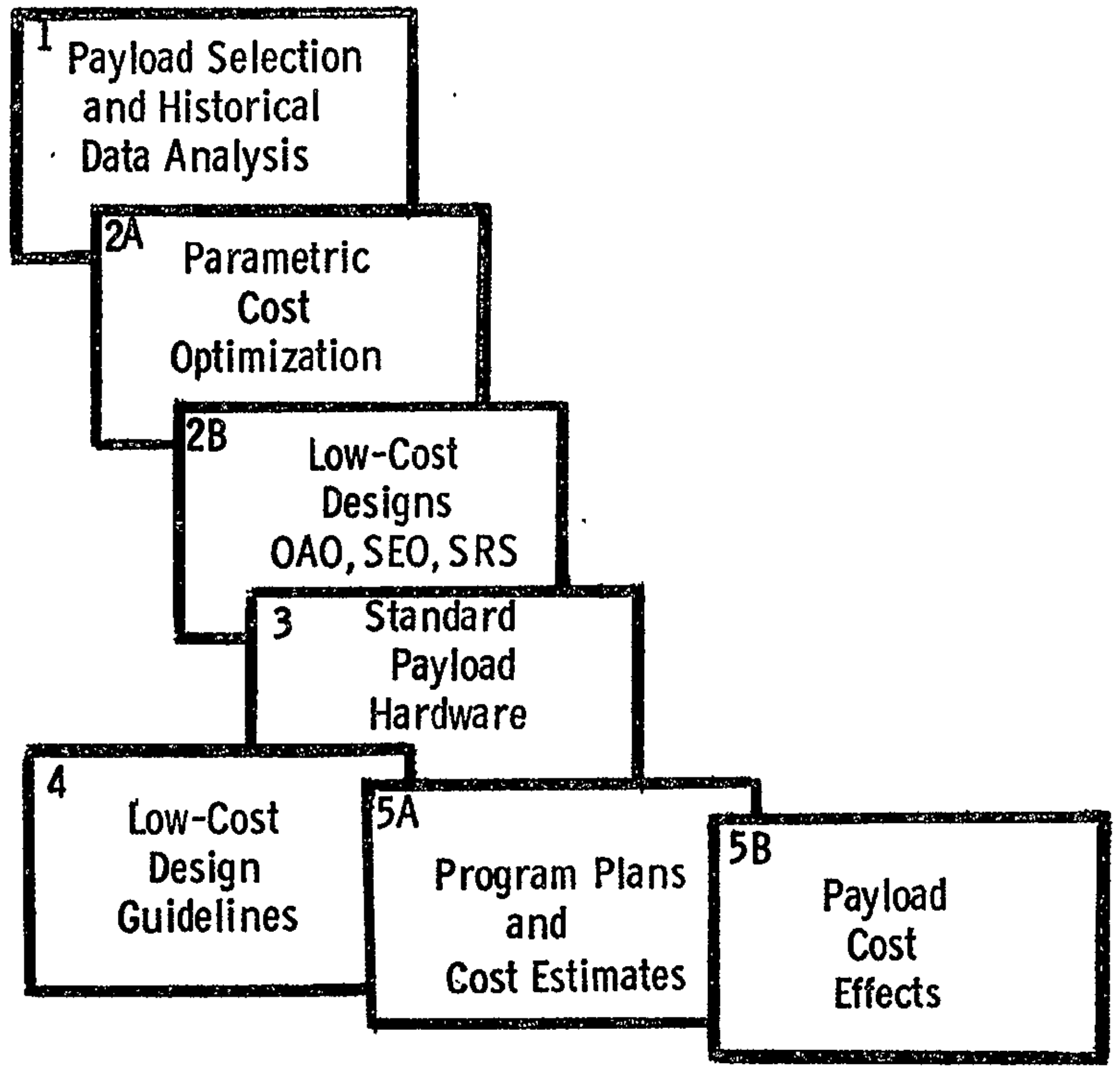
It was required for this study that payloads be selected which (1) had been flown and (2) had valid and available historical program cost data, design definition data, and operations data. Although the newer ATS, ERTS, and similar payloads might have been better candidates, otherwise, the required data were not available. The following three basic payloads therefore were selected:

- Orbiting Astronomical Observatory (OAO-B)
- Lunar Orbiter
- Small Research Satellite



LMSC ORGANIZATION - PAYLOAD EFFECTS STUDY

Fig. 2-1



STUDY TASK BREAKDOWN

Fig. 2-2

Because the Lunar Orbiter-type mission was not applicable to the new NASA mission model for the 1978-1990 time period, it was agreed with NASA that the Lunar Orbiter data would be extrapolated into two different but similar payloads, a Synchronous Equatorial Orbiter and a Mars Orbiter. The Mars Orbiter was dropped from the study after completion of initial parametric analyses. The extrapolation technique is described in Section 3 of this report. The four baseline payloads upon which the study was initially based are shown on Fig. 2-3.

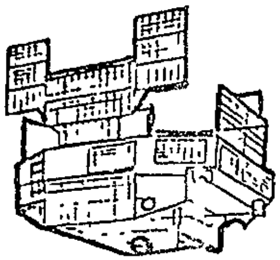
Payload Effects were expected to result from both: (1) payload hardware and program changes regardless of launch system and (2) other changes which were a function of the performance and operational environment of various launch vehicles. Aerospace Corporation specified and supplied the performance characteristics of the launch vehicle fleets which were used by LMSC in the study. The launch vehicles for each of the four baseline payloads were selected by matching the launch vehicle to the mission requirement. The combinations selected are shown on Fig. 2-4.

2.1.4 Basic Study Approach

The early study effort (a) identified the characteristics of historical and current payload programs which could be changed by the introduction of new launch/transportation systems and thereby offer cost reductions and (b) established the potential "cost-driver" payload effects.

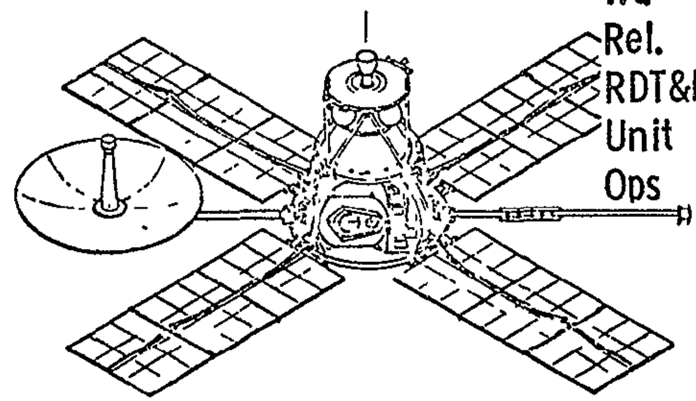
2.1.4.1 Traditional Payload Design/Operations. The philosophy which has been employed for most of the historical payloads are:

- Design within limited weight and volume constraints; high-density packaging.
- Heavy emphasis on low-risk hardware; extensive reliability and qualification testing.



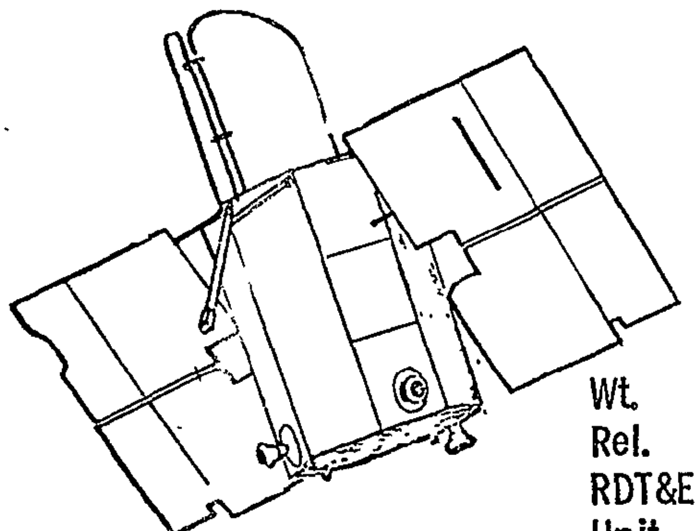
Wt.		251 lb
Rel.		.556
RDT&E	$	9.13M
Unit	$	1.39M
Ops	$	0.19M

SRS



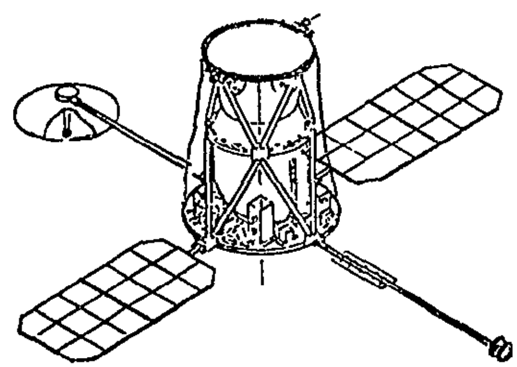
Wt.		941 lb
Rel.		.803
RDT&E	$	115.72M
Unit	$	14.66M
Ops	$	17.66M

MARS ORBITER



Wt.		4745 lb
Rel.		.609
RDT&E	$	165.41M
Unit	$	36.15M
Ops	$	11.22M

OAO



Wt.	1090 lb
Rel.	.606
RDT&E	$116.6M
Unit	$13.9M
Ops	$11.4M

SEO (2 yr.)

1 lb = 0.4536 kg

Fig. 2-3 Baseline Payloads Selected for Study

LMSC-A990556

PAYLOAD TYPE	MISSION DESTINATION	LAUNCH VEHICLE			
		Baseline	Alternate Current Expendable	New Low-Cost Expendable	Reusable STS
SRS	300 N.M. 82°	SLV-3A/ Agena	SLV-3C/ Burner II	3 Seg. SRM/ Titan Core II/ Agena	Shuttle
OAO	400 N.M. 35°	SLV-3C/ Centaur	SLV-3C/ Centaur	TIII-L2	Shuttle
SEO	Syneq. 19320 N.M.	SLV-3A/ Agena	Titan IIID/ Centaur	Titan IIID/ Centaur	Shuttle/ Space Tug
Mars Orbiter	Mars Orbit	SLV-3C/ Centaur	Titan IIID/ Centaur	Titan IIID/ Centaur	Shuttle/ Space Tug

1 N.M. = 1.852 km

SELECTED LAUNCH VEHICLE/PAYLOAD COMBINATIONS

Fig.2-4

- High level of documentation and configuration management (traceability; because failed hardware not obtainable)

- Lengthy ground checkout of payload prior to launch; large quantities of personnel on multi-shift for ground checkout, pre-launch monitoring, ascent monitoring, data acquisition, and analysis

- Project management decision required for commitment to launch (requires crew of specialists).

2.1.4.2 Influence of New Launch Vehicles. With the new expendable launch vehicles and the Space Shuttle, a new look can be taken at payload programs with the objective of simplification and reduced cost. The parameters listed following indicate the primary influences (separately) for new expendable and the Shuttle launch systems. Certain characteristics of flight attainable with the Shuttle can also be obtained with the planned expendables, but with some penalty to the latter: (a) the "softer ride" can be obtained with the new expendable but with more complexity and cost than has been planned (throttling engines, etc.); (b) also, orbit maintenance/refurbishment can be accomplished using expendables but at considerably increased technical development risk and cost.

	New Expendable	Space Shuttle
• Reduced transportation costs	X	X
• Increased weight and volume	X	X
• Space environment flight test . . .		X
• Softer ride (airplane-type operation)	(X)	X
• Payload retrieval and diagnosis . .		X
• Orbit maintenance/refurbishment. . .	(X)	X
• Checkout on orbit		X
• Intact abort		X

2.1.4.3 Principal Cost Drivers. It was determined that there were a number of principal areas in which payload program cost savings could be derived. The primary cost-drivers listed below and other cost-reduction approaches were pursued throughout the study and are explained in Sections 5, 8, and 9.

- Volume/weight limits
- Ground/flight test philosophy
- Repair/refurbishment approach
- Acceptance of risk (reliability) and payload operating life
- Quantity and quality of parts
- Use of developed/qualified hardware (off-shelf)

2.2 PAYLOAD DATA ANALYSIS AND APPORTIONMENT

2.2.1 Cost Breakdown

Cost, weight, and reliability data were obtained on each of the baseline payloads. To assure that all cost data, both for the baseline payloads and for the to-be-designed low-cost payloads, was subdivided on a directly comparable basis, a cost breakdown structure illustrated in Fig. 2-5 was established. Further, this cost-element listing was utilized as a check-list during subsequent analyses of cost-reduction potential.

2.2.2 Hardware Breakdown

To assure similar uniformity in hardware breakdown, a typical payload assembly breakdown was established. The eight subsystems are shown in Fig. 2-6. For purposes of weight, reliability, and cost tradeoffs, the experiment package was considered a subsystem and integral with the payload.

2.2.3 Apportionment of Cost, Weight, Reliability

Baseline data, as received, was not in all cases segregated into the aforementioned cost element and hardware breakdowns. Apportionment was therefore

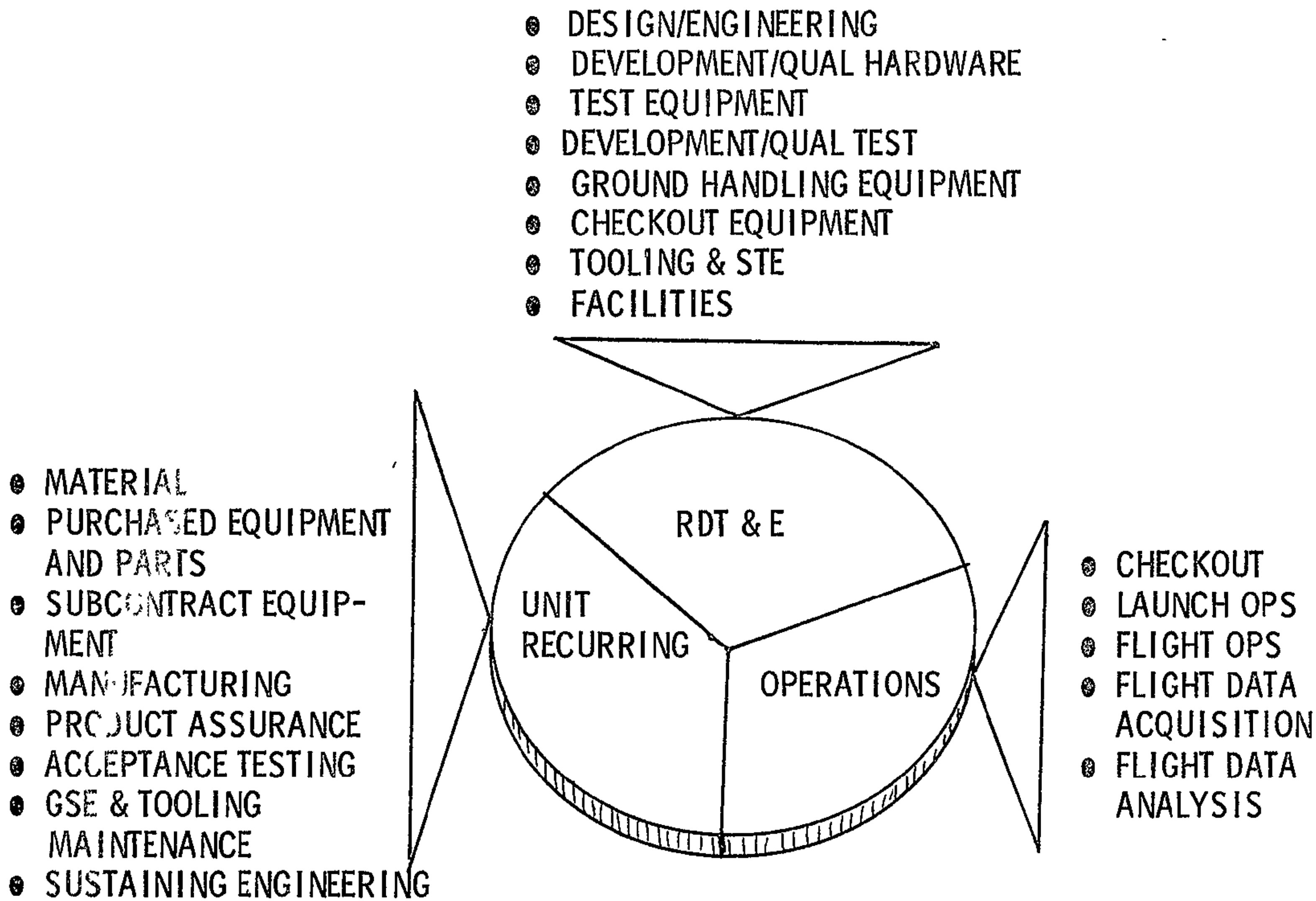


Fig. 2-5 Cost Breakdown for Visibility of Payload Effects

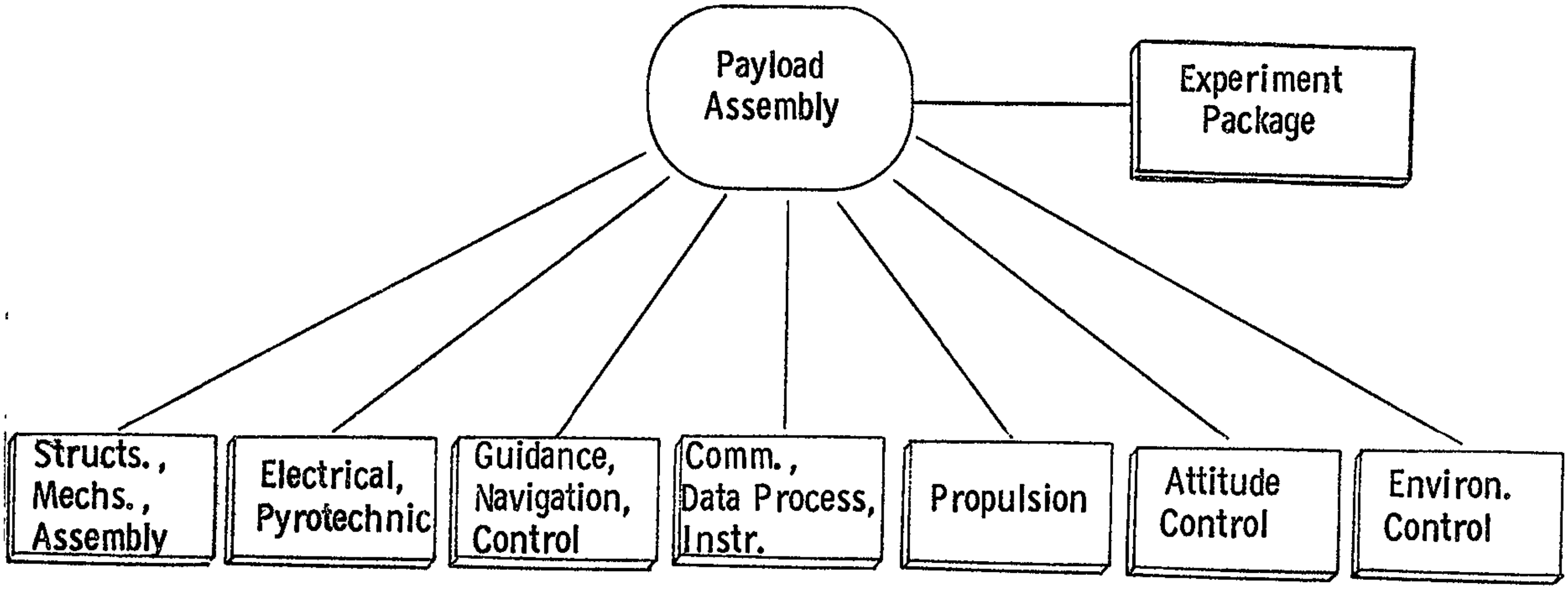


Fig. 2-6 Subsystem Breakdown for Hardware Consistency

done, using best engineering judgment and the available data, to allocate costs, weights, and reliabilities to the individual subsystems. The detail apportionments are shown in Section 3.

2.2.4 Preliminary Analysis of Payload Effects

To obtain a "feel" for the type and magnitude of cost savings potential, the cost reduction areas were matrix-plotted against the cost category affected. Figure 2-7 shows a summary of this analytical approach. In this manner, a determination was made of primary-emphasis areas for cost reduction and there was developed an early indication what savings could be derived for the Shuttle-launched versus the expendable-launched payloads.

2.3 PARAMETRIC ANALYSIS FOR INTERIM REPORT

The early need date for preliminary cost data on payload effects (7 December 1970) necessitated initiation of the parametric payload cost-optimization analysis prior to establishment of point designs for typical low-cost payloads. The baseline payload data analysis, the computerized optimization analysis, and principal results are summarized below. A detailed description is provided in Sections 3 and 4 of this report.

2.3.1 Computerized Cost-Optimization Analysis

An existing LMSC computer program was modified to: (1) accept a fairly large quantity of input data on payload, launch vehicle, and mission parameters; (2) perform a program-cost minimization calculation; and (3) re-apportion weight, reliability, and cost to each of the optimized-payload subsystems. A schematic representation of this analysis technique is shown on Fig. 2-8. One of the principal features of this cost-optimization was the use of a multi-dimensional CER (cost estimating relationship) concept which combined and traded-off the parameters of cost, weight, and reliability (including both component reliability and redundancy elements) for a constant-performance subsystem. A symbolized illustration of this concept is shown in Fig. 2-9; a reduction of

COST REDUCTION AREA	Launch Applicability	RDT&E									UNIT PRODUCTION					OPERATIONS			
		Program Mgt.	Payload Integr.	Devel/Des. Engr	Devel/Qual Hdwe	Devel/Qual Test	Test Equipment	Tooling	Checkout Equip.	Facilities	Matl./Parts	Purch. Equip.	Manufacturing	Product Assur.	Accep. Test	Checkout	Launch Ops	Flight Ops	Flt Data Acq/Anal
SIMPLIFIED CONTRACT/DOCUMENT REQUIREMENTS	E/S	X	X	X		X					X	X	X	X	X				
SIMPLIFIED CONFIGURATION MANAGEMENT (TRACEABILITY)	S	X	X	X		X					X	X	X	X	X				
USE OF PROVEN TECHNOLOGY-OFF SHELF	E/S			X		X									X				X
USE OF LOW-COST MATERIALS	E/S				X	X		X			X		X	X	X				
DECREASE STRESS LEVEL ON PARTS	E/S				X	X					X	X			X	X			
USE LOWER-QUALITY PARTS	S			X	X	X		X		X	X	X	X	X	X	X			
INCREASE STRUCTURE SAFETY MARGIN	E/S			X	X	X	X	X		X	X	X	X	X	X	X			
USE LOWER RELIABILITY GOALS (W/MAINTENANCE)	S			X	X	X	X		X		X	X		X	X	X			
INCREASE VOLUME OF PACKAGES	E/S			X				X				X	X	X					
INCREASE HARDWARE WEIGHT ALLOWANCE	E/S			X	X			X			X	X	X	X					
SIMPLIFY/MODULARIZE HARDWARE	E/S			X	X	X	X		X		X	X	X	X	X	X			X
SIMPLIFY/REDUCE GROUND TESTING	S	X	X	X	X	X	X			X									
USE ORBIT MAINTENANCE/REFURBISHMENT AND REUSE	S	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
EMPLOY HARDWARE UPDATE (VS NEW HARDWARE)	S	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
USE PRE-DEPLOYMENT ORBIT CHECKOUT	S			X	X	X	X		X	X	X	X		X	X	X		X	X

* E = EXPENDABLE; S = SHUTTLE

Fig. 2-7 Potential Cost Reduction Areas

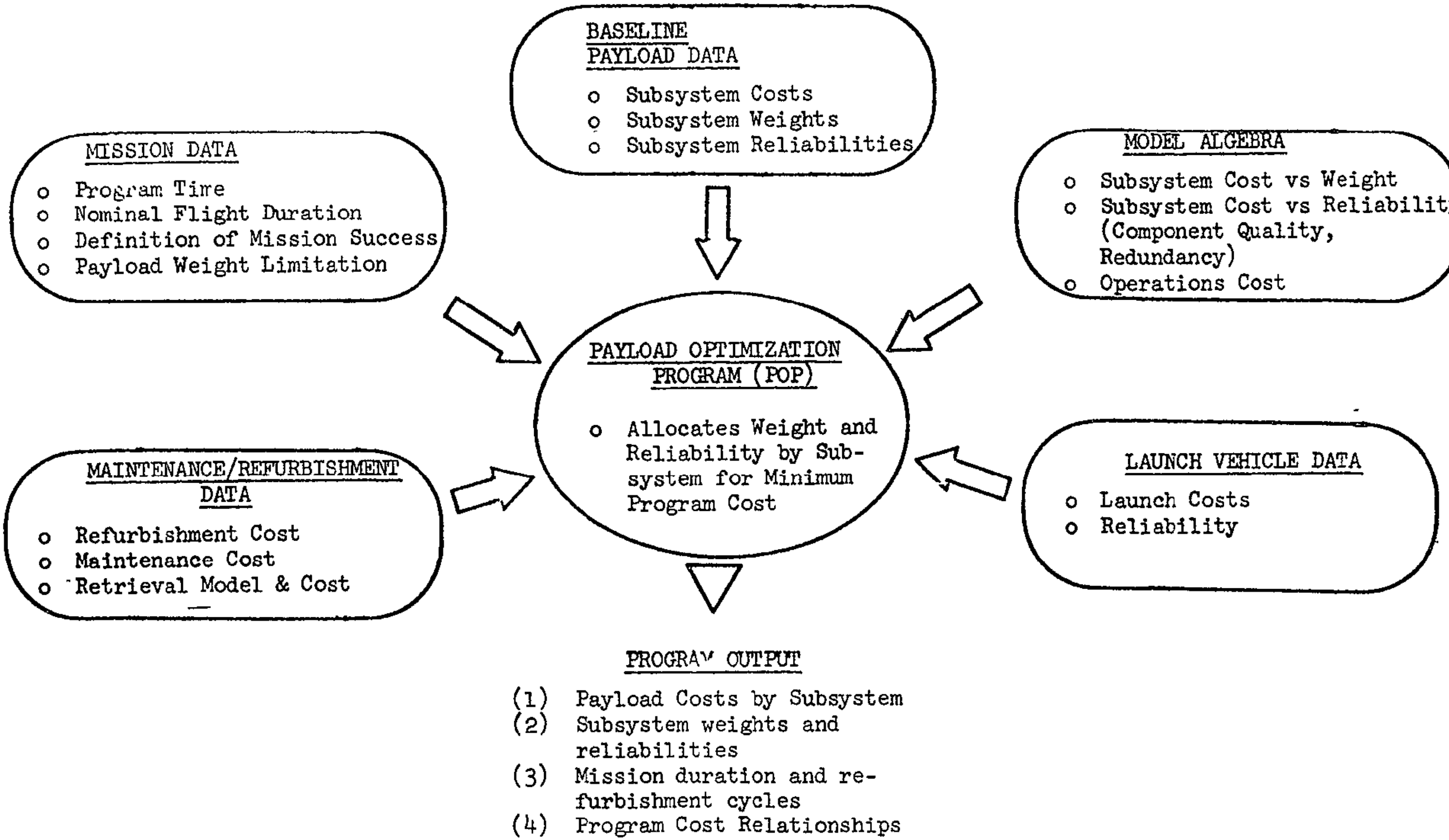


Fig. 2-8 Cost Optimization Analysis for Payload Programs

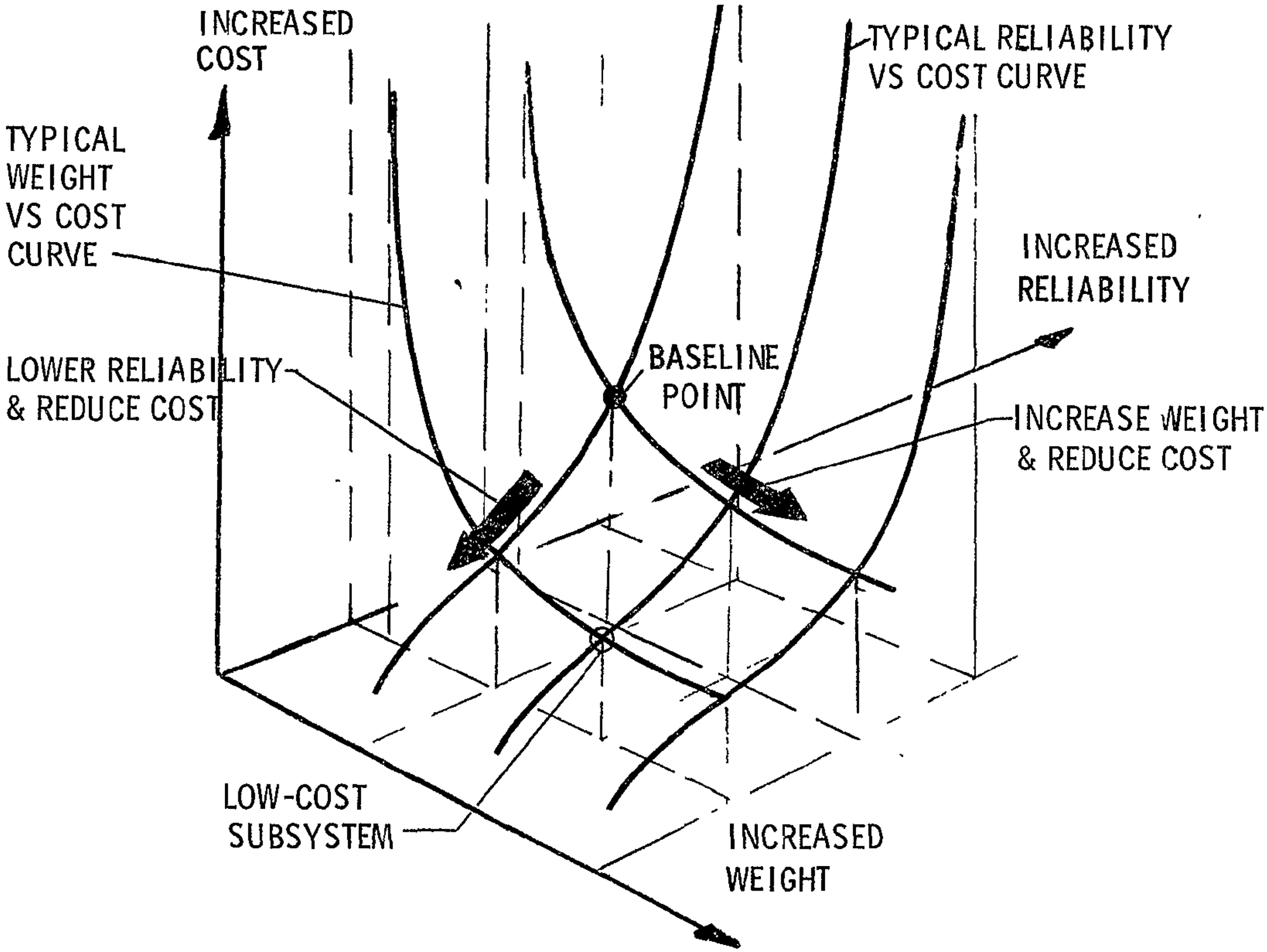


Fig. 2-9 N-Dimensional CER - Typical Subsystem

required reliability, an increase in weight, or a combination of these is indicated to decrease the cost of a subsystem below that of the baseline. The development and use of this concept is described in Section 4.

A total of 69 computer runs (the complete matrix is shown in Section 4) were made for the four payloads (OAO, SEO, SRS, and Mars Orbiter) used in combination with the three types of launch vehicles. The input variants included:

- Payload weight limits
- Program time
- Refurbishment cycle and cost ratio
- Launch cost

The significant results of the parametric cost-optimization analysis were:

- Payload cost savings using Shuttle are significant
- Payload cost savings using new-expendable systems are attainable but less than for Shuttle-launched
- Periodic refurbishments and reuse (with Shuttle system) provides a principal program cost saving
- The tradeoff/selection of refurbishment vs payload life/reliability is strongly influenced by:
  - Payload vs launch vehicle cost ratio
  - Refurbished vs new payload cost ratio

2.3.2 Interim Report Data

Preliminary data based on the parametric analysis was supplied to NASA and to Aerospace Corporation on 7 December 1970, and later validated and expanded in the LMSC "Interim Report - Payload Effects Analysis", LMSC-A983808 dtd 22 December 1970. These data included those items listed on Fig. 2-10.

- BASELINE PAYLOAD COST DATA BY SUBSYSTEM
- BASELINE PAYLOAD WEIGHT, AND RELIABILITY DATA BY SUBSYSTEM
- LOW-COST PAYLOAD TARGET COSTS (OAO, SEO, SRS, MARS ORBITER) BY SUBSYSTEM AND BY COST CATEGORY
- LOW-COST PAYLOAD WEIGHT, VOLUME, AND RELIABILITY CHARACTERISTICS
- PARAMETRIC PAYLOAD EFFECTS ON PROGRAM COSTS
  - REFURBISHMENT CYCLE TIME AND COST
  - MISSION DURATION AND PAYLOAD LIFE REQUIREMENTS
  - RELIABILITY AND WEIGHT SENSITIVITY

Fig. 2-10 Mid-Term Data Drop to Aerospace

The summary of the preliminary costs or "cost targets" for the OAO, SEO, and SRS used with the Space Shuttle are shown on Fig. 2-11. Cost target data for the expendable-launched payloads are included in Section 4.

Supplementing the cost data, preliminary engineering estimates of the weight and volume (envelope configuration) were made of the four payloads to aid Aerospace Corporation in their "capture analysis". These data were included in document LMSC-A973883 dtd 25 November 1970. Because the 2-year SEO was developed at a later date, an estimated weight was not available at the time of the interim report.

2.4 DESIGN OF LOW-COST PAYLOADS

A low-cost version of each of the baseline payloads, OAO, SEO, and SRS, was designed. The following is a brief resume of the groundrules established and designer indoctrination which preceded the design, the results of the low-cost design effort, and the development of guidelines for future payload design. A considerable amount of detail is provided in Section 5 of this report.

2.4.1 Initial Groundrules for Low-Cost Design

As a first step, the complete set of baseline design data for each payload was thoroughly reviewed to understand the relative complexity of each subsystem, the parts and components used, and the type and amount of testing which had been performed. The payload subsystem characteristics were then evaluated relevant to the potential for cost reduction. A sample matrix is shown in Fig. 2-12 for the OAO.

Instructions were then given to the designers to: (1) familiarize them with the results of the parametric cost-optimization analysis; (2) explain basic low-cost design approaches; and (3) illustrate the effect of various design approaches upon program costs. The principal instructions are listed on Fig. 2-13. In addition, the following basic groundrules were established:

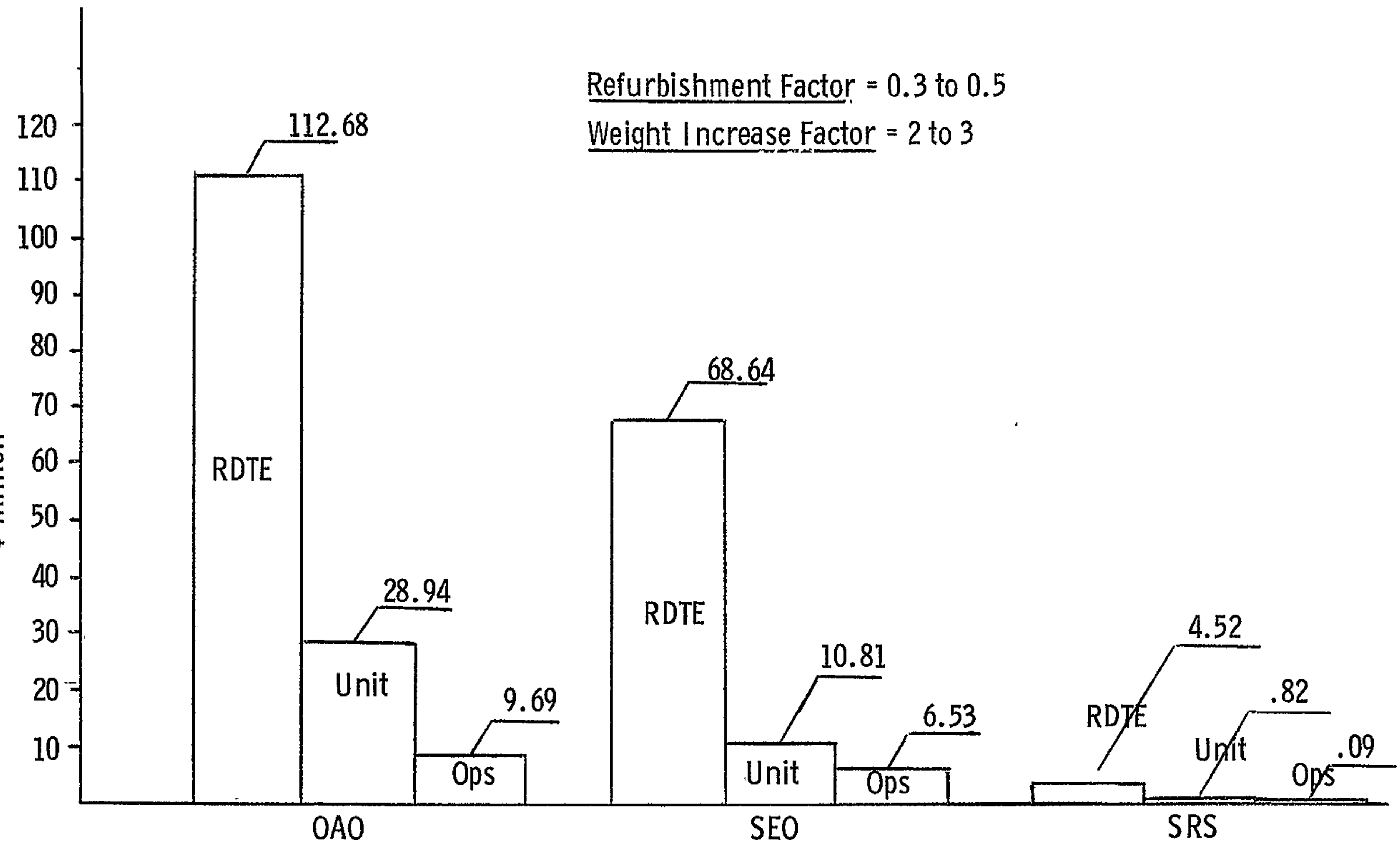


Fig. 2-11 Low-Cost Payload Cost Targets (Shuttle-Launched)(Used for Interim Data Reports)

PAYLOAD TYPE - OAO	LAUNCH VEHICLE - SPACE SHUTTLE													
	PAYLOAD COST REDUCTION POTENTIAL BY CHARACTERISTIC*													
SYSTEM	VOLUME	WEIGHT	RELIABILITY	MISSION TIME	MATERIALS	COMPONENTS	DESIGN SAFETY FACTORS	C/O REQUIREMENTS & LAUNCH SUP'T	HDLG./SERVICING REQUIREMENTS	FAULT ISOLATION CAPABILITY	ON-ORBIT MAINTAINABILITY	RETRIEVABILITY	REUSE/REFURB. CAPABILITY	MISSION OPS. SUPPORT
PAYLOAD ASSEMBLY & INTEGRATION	H	H	L	L	L	L	H	L	M	L	L	L	L	L
EXPERIMENTS	H	H	M	M	H	H	M	H	L	L	H	H	H	H
STRUCTURES AND MECHANISMS & S/C ASSEMBLY	H	H	M	L	M	H	H	L	L	L	L	L	M	L
ELECTRICAL AND PYROTECHNICS	H	H	H	H	M	H	M	L	L	L	M	M	M	M
GUIDANCE, NAVIGATION, STABILITY, CONTROL	H	H	H	H	M	H	M	H	L	H	H	H	H	H
PROPULSION AND ATTITUDE CONTROL	M	M	H	H	H	H	H	M	L	L	M	L	L	L
TELEMETRY, TRACKING AND COMMUNICATION	H	H	H	H	M	H	M	H	L	H	H	H	H	H
ENVIRONMENTAL CONTROL	L	M	L	L	L	L	L	L	L	L	L	L	L	L

* H = HIGH, M = MODERATE, L = LOW

Fig. 2-12 Priority of Cost-Reduction Potential (Example)

- REVIEW PARAMETRIC TARGET COSTS AND CONSIDER RELATIONSHIPS AND COST EFFECTS OF WEIGHT, RELIABILITY, AND REFURBISHMENT
- REPETITIVE REVIEW OF PAYLOAD DESIGN EFFECTS UPON PRIMARY COST-DRIVER ELEMENTS
- RDT&E COST REDUCTIONS PREFERRED OVER UNIT COST REDUCTION
- USE PROVEN TECHNOLOGY AND QUALIFIED OFF-SHELF HARDWARE
- MINIMIZE DEVELOPMENT/QUALIFICATION TESTING
- REDUCE TYPES AND QUANTITY OF DIFFERENT COMPONENTS
- PROVIDE MAXIMUM ACCESSIBILITY
- PROVIDE FOR GROWTH AND UPDATING

Fig. 2-13 Special Instructions to Designers

- Volume and Weight - Minimum constraints
- Performance - Configuration, functions, and hardware may be altered to obtain cost reduction; however, overall performance capability must be retained.
- Overdesign - Use high structural safety factors and reduce parts/component stress levels
- Modularization - Equipment to be modularized to facilitate on-orbit replacement and refurbishment/reuse
- Hardware Complexity - Reduce without affecting overall payload reliability
- Materiel - Use inexpensive materials, off-shelf components

For the Shuttle-launched payloads, a design premise was developed that distinguished "man-safety" from "man-rating". In agreement with the Technical Monitoring Team, it was determined that the man-safety requirements as listed on Fig. 2-14 should apply.

2.4.2 Specifications for Low-Cost Payloads

LMSC prepared a design/performance specification for each of four payloads:

OAO	LMSC-A973890
SEO	LMSC-A981600
SRS	LMSC-A981647
Mars Orbiter	LMSC-A984063

A summary sheet listing some of the basic specification requirements for the low-cost OAO is shown on Fig. 2-15.

MAN-RATING OF PAYLOADS (NOT REQUIRED)

FAILURE OF PAYLOAD HARDWARE USUALLY DOES NOT CONTRIBUTE DIRECTLY TO SAFETY OF FLIGHT. IT NEED NOT BE MAN-RATED. EXAMPLE: COMPLETE FAILURE OF PAYLOAD ELECTRICAL SUBSYSTEM HAS NO EFFECT ON SHUTTLE SAFETY OF FLIGHT.

MAN-SAFETY

ALL PAYLOAD HARDWARE ELEMENTS, FAILURE OR MALFUNCTION OF WHICH MAY DAMAGE THE SHUTTLE OR INJURE PERSONNEL, MUST BE DESIGNED TO PREVENT THE FAILURE OR TO RESTRAIN THE EFFECTS WITHIN THE PAYLOAD:

- USE HIGHER SAFETY FACTORS AND OVERDESIGN
- FOR FLUID SYSTEMS, DESIGN TANKAGE TO WITHSTAND OVERPRESSURES EVEN IF RELIEF VALVES FAIL
- PROVIDE BLAST SHIELDS FOR EXPLOSIVE DEVICES
- PROVIDE DEACTIVATION WHEN CREW IS NEAR PAYLOAD

Fig. 2-14 Man-Rating vs Man-Safety

- SPEC. NO.: LMSC-A973890, dated 4 Dec 1970
- PAYLOAD NOMINAL LIFE
  - (a) 1-Year (Hi-Rel parts) - Reliability Goal = .609 (Experiment = .940; Spacecraft = .648)
  - (b) 4-8 month (lower-quality parts)
- MAINTENANCE/REFURBISHMENT:
  - (a) Consider operation up to 5-years with periodic refurbishment at 1-yr intervals
  - (b) Consider maintenance and experiment update at 4 to 8 month intervals
  - (c) Docking with Space Tug and/or Shuttle
- LAUNCH VEHICLES:
  - (a) Space Shuttle - 30,900 lb to OAO orbit (400 nm; 35 deg)
  - (b) New Low-Cost Expendable - TIII-L2 - 10,000 lb to OAO orbit
  - (c) Alternate Current Expendable - Atlas/Centaur - 9500 lb to OAO orbit
- EXPERIMENT:

  Stellar Telescope - 38 in. aperture with 1100 to 4267-A spectral range; and
  Spectrometer 1 arc sec pointing accuracy
- OPERATIONAL ORBIT:
  - (a) 390 to 417 nm circular at 35 deg - orbit period 101 minutes
  - (b) 348/520 nm elliptical at 35 deg

1 lb = 0.4536 kg
1 in = 2.54 cm
1 nm = 1.852 km

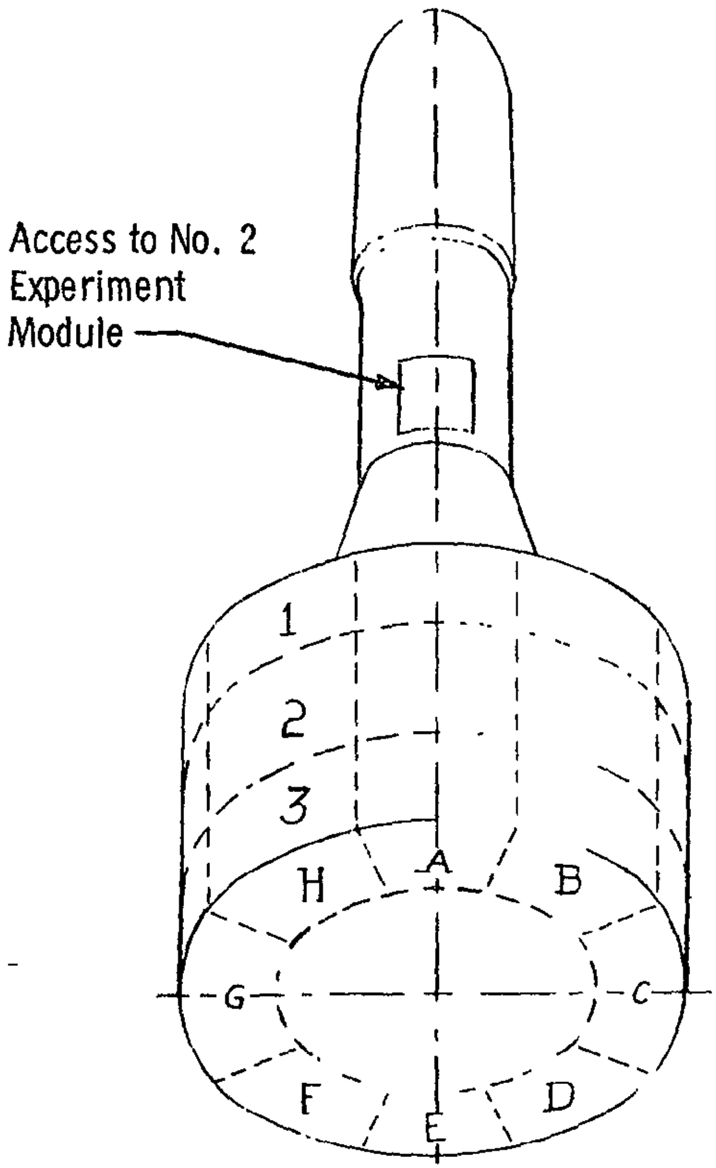
Fig. 2-15 Low-Cost OAO Requirements Highlights

2.4.3 Modular Design of Low-Cost Payloads

Because of the prime importance of modular design to allow in-orbit replacement of payload equipment for repair or refurbishment, explicit requirements were established for the OAO and SEO. Following is a summarization:

- Divide payload subsystems into minimum quantity of modules consistent with:
    - Maximum weight/size which can be readily installed or removed by space crew
    - Maximum cost of a single module which is economical for spares replacement
- Segregate components which have high probability of replacement from those which have higher predicted life.
- Establish operating tolerances on individual modules so that module replacement will not require payload recalibration.
- Provide simple functional and mechanical interfaces between modules.
- Provide for easy access to and removal/installation of modules without need for special tools.

The modular design was actually implemented on the OAO and the SEO as schematically illustrated in Figs. 2-16 and 2-17. It may be noted that some compartments have been left empty to accommodate future growth and/or update. Figure 2-18 illustrates the four different modules of the electrical power subsystem for the low-cost SEO. In most cases, the modules of the various subsystem are a common size, approximately 14 x 24 x 30 in. (36 x 61 x 76 cm). As is pointed out in Section 6, the module approach accrues cost savings in manufacturing assembly and testing as well as providing ease of equipment replacement and refurbishment.

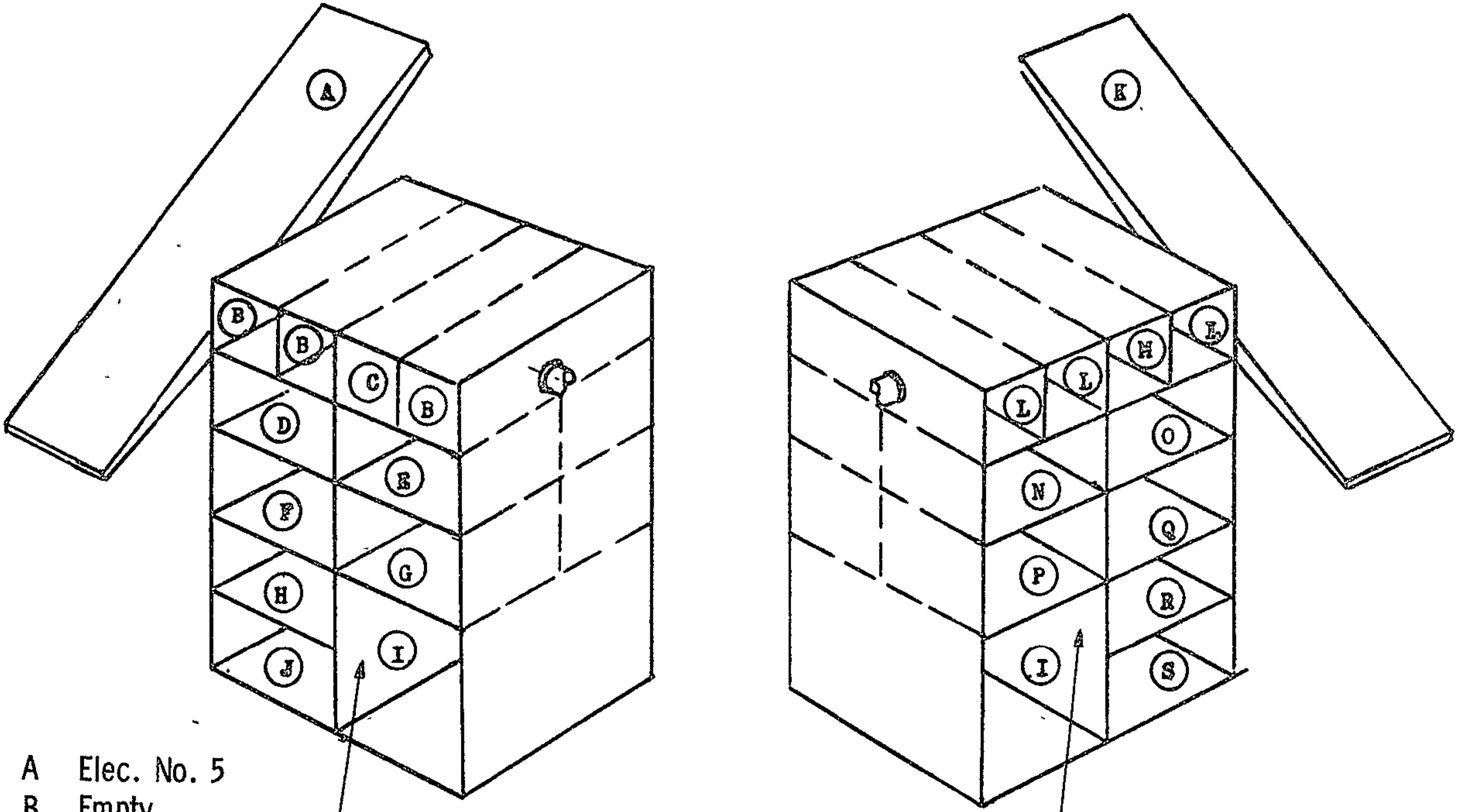


Compartment	Module
A-1	Empty
A-2	Electrical Power No. 2
A-3	Attitude Control No. 1
B-1	Empty
B-2	Electrical Power No. 1
B-3	Empty
C-1	Empty
C-2	Empty
C-3	Attitude Control No. 2
D-1	Stabilization & Control No. 1
D-2	Experiment No. 1
D-3	Empty
E-1	Stabilization & Control No. 3
E-2	CDPI*Data Distribution Unit**
E-3	Attitude Control No. 3
F-1	Stabilization & Control No. 2
F-2	CDPI* No. 1
F-3	Empty
G-1	Empty
G-2	CDPI No. 2
G-3	Attitude Control No. 4
H-1	Empty
H-2	Stabilization & Control No. 4
H-3	Empty

* Communications, Data Processing, & Instrumentation
** Fixed unit, not replaceable in orbit

Fig. 2-16 Low-Cost OAO Module Locations

LMSC-A990556



A Elec. No. 5
B Empty
C Elec. No. 3
D Elec. No. 1
E CDPI No. 1
F ACS No. 1
G ACS No. 2
H CDPI No. 3
I Exper. No. 1
J Exper. No. 2

K Elec. No. 6
L Empty
M Elec. No. 4
N CDPI No. 2
O Elec. No. 2
P ACS No. 3
Q ACS No. 4
R S&C No. 1
S S&C No. 2

Fig. 2-17 SEO Module Location & Arrangement

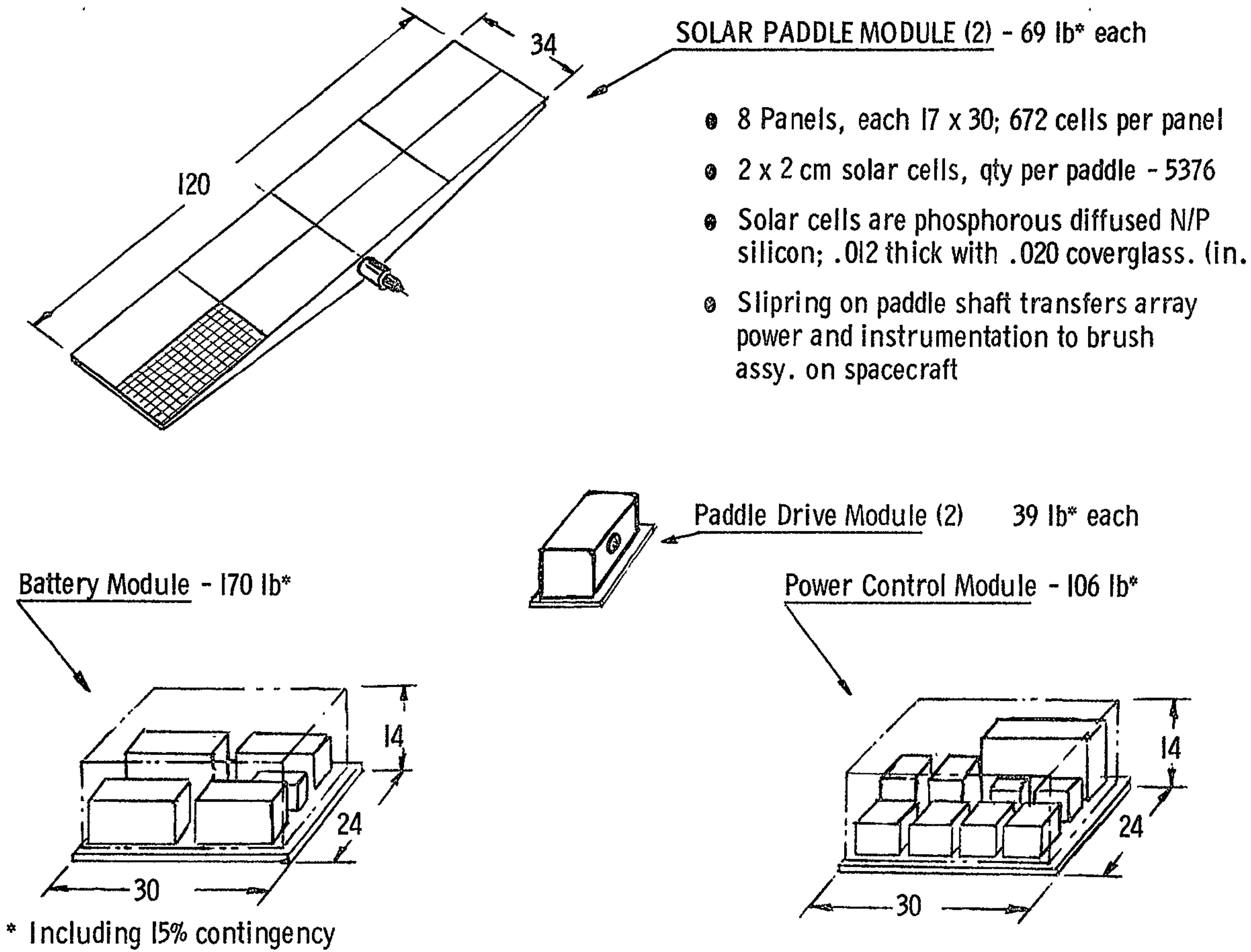


Fig. 2-18 Low-Cost SEO Electrical Subsystem

2.4.4 Parts and Component Reliability

It was recognized early in the study that reduction in payload part/component quality might be compatible with the orbit revisit capability of the Shuttle for repair or refurbishment of the payloads. Investigations were made into relative hardware costs for various quality grades and the practicability of using the lower-cost parts on future payloads. The mission model requirements covering the Space Shuttle operational time period indicated a need for high-reliability parts and components. Therefore, the lower-cost (MIL-Spec and aircraft) parts were not pursued to their final potential. This area should be studied in more depth as mission equipment and spacecraft subsystems become better defined.

2.4.4.1 Percentage of Payload Cost Allocated to Parts/Components. The percentage of the payload recurring cost which is allocatable to purchased parts and components ranges from about 10 to 20 percent, depending upon how much hardware is "off-the-shelf" versus how much is in-house special "make" category. The parts and components used in the baseline payloads studies were all hi-rel type and therefore of highest cost.

2.4.4.2 Comparison of Part Cost vs Quality. The parts investigated were in four basic categories; these categories, their description, and the comparative price ranges are shown on Fig. 2-19.

2.4.4.3 Comparison of Failure Rates and Weights. The failure rate of MIL-Spec parts is about 2 to 3 times that of hi-rel parts. The failure rate of aircraft parts is about 10 times that of hi-rel parts, but more importantly, the drift rate is about 4 times the hi-rel part drift rate. In general, the MIL-Spec parts are comparable in weight to the hi-rel; the aircraft-equivalent parts are noticeably heavier, as indicated by a sample listing on Fig. 2-20.

2.4.4.4 Application Analysis of MIL-Spec Parts. A preliminary analysis was made of MIL-Spec part application to an SEO-type payload. It was determined feasible to use these parts in lieu of hi-rel if the design operating life of

HI - RELIABILITY - COST = \$x

PRODUCED ON SPECIAL PRODUCTION LINES IN MOST CASES; MANY PARTS 100% TESTED FOR PERFORMANCE; ALL PARTS SCREENED AND BURNED IN; LIFE OF 2-3 YEARS WITH GOOD PARAMETRIC STABILITY IS POSSIBLE; FAILURE RATES VERY WELL DOCUMENTED

MIL-SPEC - COST = \$0.3x TO \$0.5x

SAMPLE SELECTED FROM PRODUCTION RUN FOR TEST TO SPEC. REQTS. - SAMPLE PASSING TEST WILL ACCEPT TOTAL LOT; FAILURE RATE 2x TO 3x HI-REL; APPROX. 70% OF HI-REL PARAMETRIC STABILITY

AIRCRAFT - COST = \$0.15x TO \$0.2x

PARTS TESTED TO PARAMETRIC NORMS; SAMPLE FROM EACH LOT SUBJECTED TO SIMULATED STRESSES AND SHOCKS EXPECTED IN AIRCRAFT FLIGHT - SAMPLE FAILURE REJECTS TOTAL LOT; FAILURE RATE ABOUT 10x HI-REL PARTS; DRIFT RATE ABOUT 4x HI-REL

COMMERCIAL - COST = \$0.05x TO \$0.1x

MANUFACTURED IN VERY LARGE QUANTITIES; SUBJECTED TO SIMPLE GO-NO-GO TESTS, SAMPLING TECHNIQUES TO ACCEPT/REJECT LOTS; NO FAILURE RATE DATA EXCEPT BY LOTS REJECTED

Fig. 2-19 Categories of Parts/Components

PART OR COMPONENT	COST (APPROXIMATE $ PER UNIT)			
	Commercial	Aircraft	Mil-Spec	Hi-Rel
Pressure Transducer	1.65	4.00	10.00	65.00
Transistor	0.25	0.75	2.50	7.50
Resistor	0.17	0.60	2.20	4.90
Coax Cable RG 58U	0.14/ft.	0.35/ft	1.15/ft	2.40/ft
Capacitor (Ceramic cap)	0.24	0.47	1.60	4.85
Connector (16 pin)	0.60	1.35	3.15	8.00
Integrated Circuit (and Gate and Paramp)	1.15	2.75	7.00	16.60

Fig. 2-20 Comparison of Cost for Different Quality Parts

the SEO (time between refurbishments) could be limited to 9 months. Further analysis of this part application approach was not pursued further because of the higher priority of completing the low-cost 2-year SEO preliminary design and analysis, which was more representative of the predicted future mission requirements.

2.4.4.5 <u>Application Analysis of Aircraft Parts</u>. Another preliminary analysis was made of substituting aircraft quality parts on the OAO in lieu of hi-rel parts. Although it was determined to be feasible, the operating life of the OAO was reduced to 4-months, principally as a result of the relatively high drift rates of the aircraft parts. Further, and more detailed, analysis of use of aircraft-quality parts would be worthwhile if:

a. It was otherwise economically feasible to revisit and refurbish an orbiting payload at short time intervals, such as 4 months.

b. The additional weight penalty of the aircraft parts was found to be tolerable (for the OAO, an additional weight of about 2,000 lb was estimated for use of aircraft-type parts versus hi-rel).

With the short-duration sortie missions planned with the Shuttle, the use of aircraft parts might show significant benefits. Further study should be undertaken of this low-cost part application as the sortie mission hardware requirements are firmed-up.

2.4.5 <u>Low-Cost Subsystem and Payload Designs</u>

Each subsystem of the three candidate payloads, OAO, SEO and SRS, was analyzed as to functional efficiency and general cost-effectiveness of the hardware. Low-cost design methodology was applied and a substitute low-cost subsystem was developed. The design outputs included:

- Parts lists with weight breakdown
- Block diagram and functional description of subsystem

- Dimensions and alignment/calibration requirements
- Special test requirements
- Approximate component costs (where known)
- Special capabilities in ground and flight operations
- Special interface requirements with launch vehicle

These data were documented in a number of LMSC engineering memoranda which were provided to NASA agencies and Aerospace Corporation for information and comment; they are listed following:

Subsystem or Vehicle	OAO	SEO	SRS
Experiments	PE-1	PE-21	PE-41
Stabilization & Control	PE-2	PE-22	PE-42
Communications, Data Processing & Instrumentation	PE-3	PE-23	PE-43
Electrical	PE-4	PE-24	PE-44
Attitude Control & Propulsion	PE-5	PE-25	PE-45
Environmental Control	PE-6	PE-26	PE-46
General Description of Payload-Shuttle-Launched	PE-7	PE-27	PE-47
General Description of Payload-Expendable-Launched	PE-8	PE-28	PE-48

2.4.5.1 Low-Cost Subsystem Characteristics. A summary listing of the principal cost-reduction features of the low-cost subsystems is shown in Fig. 2-21. Special attention was devoted to investigating methods for cost-reduction in electronic assemblies. After review of historical design, manufacturing, and product assurance at LMSC on a large variety of electronic flight hardware, basic low-cost design principles were established; Fig. 2-22 is a summary listing of these.

2.4.5.2 Low-Cost Payload Configurations. A structural design has been developed for each payload, with external configuration being determined by the volumetric need for equipment module mounting. The general configurations of the three low-cost payloads are illustrated in Figs. 2-23, 2-24, and 2-25.

STRUCTURES & MECHANISMS

- Low-cost materials, manufacturing processes
- Simple structure, high safety factors (3 or more)
- Maximum-allowable dimensional tolerances
- Eliminate extension mechanisms where possible

AVIONICS

- Increase on-board data processing capability (computer)
- Utilize Shuttle GNC capability for initial orbit positioning
- Design modules, components to allow replacement without recalibration
- Reduce packaging density
- Standardize hardware elements (circuits, PCBs, etc.)

EXPERIMENTS

- Standardized and versatile interface with payload
- Design for fixed-mounting and ground alignment where possible
- Mechanisms to be self-supporting in 1-g field

ELECTRICAL

- Ruggedized simple sheet metal structure for solar array structure
- Fixed solar arrays - eliminate folding where possible
- Large-size, 97.5% yield solar cells
- Design for long-time degradation to reduce average refurbishment cost

Fig. 2-21 Principal Contributors to Subsystem Cost Reduction

- STANDARDIZE BOX AND PCB SIZES AND CONNECTORS WHERE POSSIBLE
- STANDARDIZE CIRCUITRY ELEMENTS
- DECREASE CIRCUIT DENSITY OF PARTS - REDUCTION FROM 75% TO 30% ALLOWS ABOUT 35% SAVING IN MANUFACTURING/INSPECTION LABOR
- DESIGN THE BREADBOARD UNIT AS A PRODUCTION PROTOTYPE - USE PRODUCTION PROCESSES FOR FABRICATION OF BREADBOARD
- INCREASE CONDUCTOR SPACING; PROVIDE "BLANK" SPACE ON PCBs
- USE CONFORMAL COATING IN LIEU OF HARD POTTING TO ALLOW REPAIRS
- USE PCBs IN LIEU OF CORDWOOD MODULES - ALLOWS REDUCTION UP TO 50% OF TROUBLE-SHOOTING AND INSPECTION LABOR

Fig. 2-22 Some Approaches to Design of Low-Cost Electronic Boxes

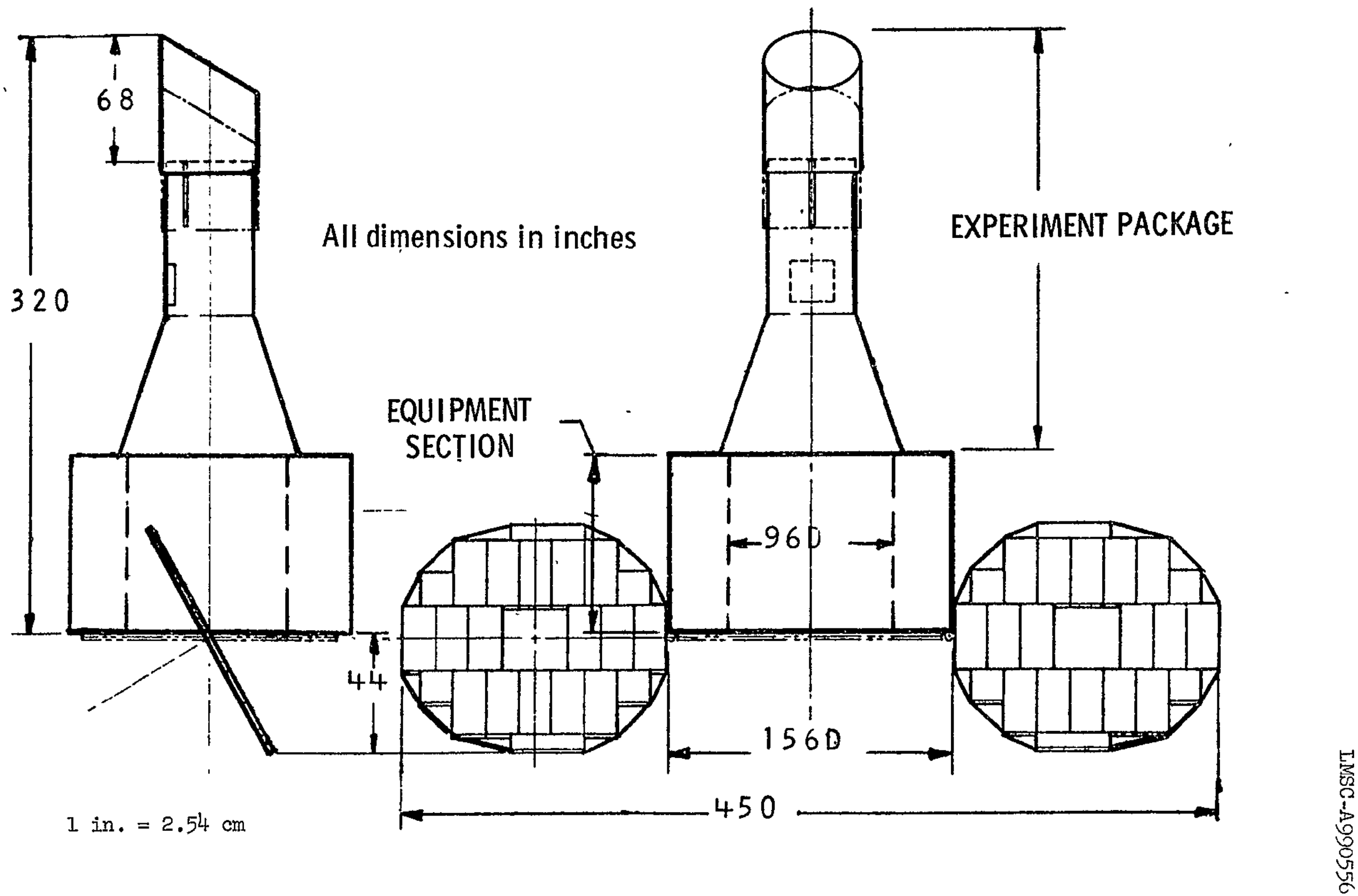


Fig. 2-23 Low-Cost OAO Configuration

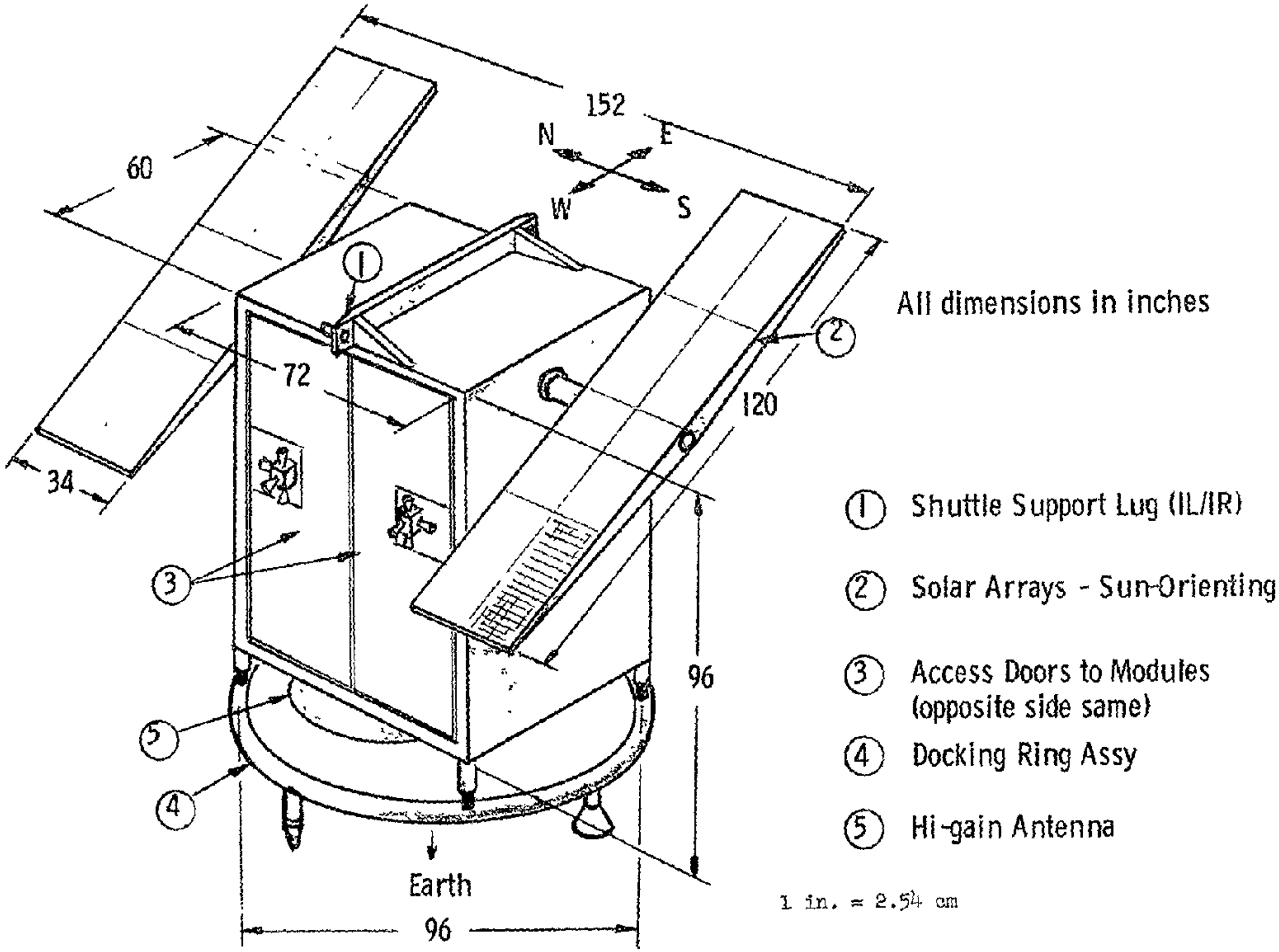


Fig. 2-24 Low-Cost SEO General Configuration

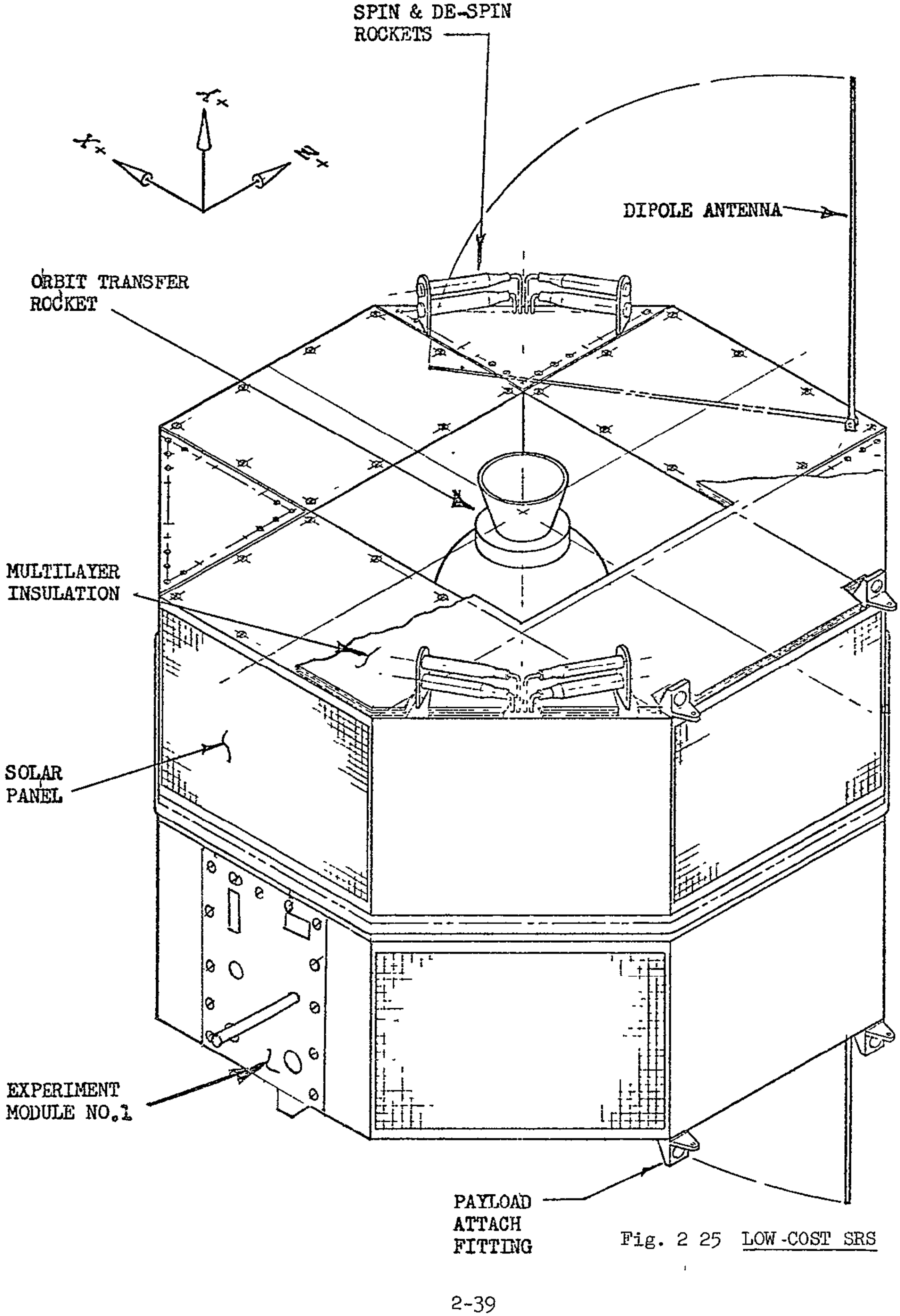


Fig. 2 25 LOW-COST SRS

2.4.5.3 Weights of Low-Cost Payloads. Weight statements were prepared for the OAO and SEO comparing the baseline with the low-cost versions are shown on Figs. 2-26 and 2-27.

2.5 PLANS AND COST ESTIMATES

To provide the basis for estimating the costs of developing, manufacturing and operating the low cost payloads, program plans were prepared for each of the three payloads. Using these plans and the design data (described in detail in Section 5), bottom-up cost estimates were made. The various details of the planning and costing effort are included in Section 6 of the report. Following is a summary of the highlights.

2.5.1 Planning Approach

Program plans were prepared covering development, qualification, manufacturing, and operations of the low-cost payloads. The basic guidelines used are listed on Fig. 2-28. A typical master schedule developed for a low-cost payload program (OAO-B) is shown on Fig. 2-29.

2.5.2 Cost Estimates - Low-Cost Payloads

Cost estimates were made on each low-cost payload program. The basic approach used is summarized on Fig. 2-30.

The summary of RDT&E, unit, and operations cost for each Shuttle-launched payload is shown on Fig. 2-31; the baseline costs are shown for comparison. The OAO figures for designs with and without a computer are shown. (Par. 2.5.4 explains the computer significance.)

2.5.3 Recosting of Baseline OAO, SEO

Because of the desire to have a "calibration" of the LMSC estimates so that true delta-cost values could be derived between the low-cost payloads and the

SUBSYSTEM	BASELINE OAO WEIGHT (LB)	LOW-COST OAO WEIGHTS (LB)*	
		SHUTTLE-LAUNCHED	EXPENDABLE** BOOSTER-LAUNCHED
EXPERIMENT	967	1,970	1.985
STRUCTURE AND MECHANISMS	1,141	1,762	1,787
STABILIZATION AND CONTROL	716	655	726
COMMUNICATION, DATA PROC., INSTRUMENTATION	456	443	457
ELECTRICAL	1,232	1,775	1,859
ENVIRONMENTAL CONTROL	100	100	100
ATTITUDE CONTROL	199	883	883
PAYLOAD TOTAL (DRY)	4,811	7,588 ***	7,797 ***

* Includes approximately 15 percent contingency.
** Also requires a payload adapter weighing 291 lb.
*** Add 320 lb of Freon for total payload weight

1 lb = 0.4536 kg

Fig. 2-26 Baseline and Low-Cost OAO Weights

HARDWARE ELEMENT	BASELINE SEO	SHUTTLE-LAUNCHED LOW-COST SEO**	EXPENDABLE-LAUNCHED LOW-COST SEO**
Experiment Package*	294 lb	518 lb	518 lb
Structures & Mechanisms	133	742	722
Electrical Power	312	580	580
Attitude Control	70	573	573
Stabilization & Control	136	223	223
Communications, Data Processing, & Instrumentation	147	254	277
Environmental Control	11	73	73
Total Dry Weight	1091 lb	2963 lb	2966 lb
Attitude Control Gas (Freon 14)	60	164	164
Total Payload Weight	1151 lb	3127 lb	3130 lb ***

* Including 12 lb N_2
** Including weight contingency of approx. 15%
*** Adapter weighing 265 lb. also required

1 lb = 0.4536 kg

Fig. 2-27 Weight Summary - Low-Cost SEO

LMSC-A990556

2-42

- COMPARABILITY TO BASELINE RETAINED
  - No refurbishment costs included
  - Flight-article quantity same as baseline
  - Equipment and software development comparable to baseline program
- NASA PHASED PROJECT PLANNING APPROACH

  OAO - 4 1/2 Year Program - Phase B to launch
  SEO - 3 1/2 Year Program - Phase B to launch - combined Phase C/D (Lunar Orbiter)
- CURRENT NASA/DOD PROGRAM MANAGEMENT APPROACHES
  - MIL-STD-499
  - NHB 5300.4 (1A) and 1B)
- GFE ASSUMED
  - Launch vehicle, fairings, and adapters, launch services control center, STADAN, NASCOM, operational computer
- SIMILAR SPACECRAFT DESIGN FOR EITHER SHUTTLE OR EXPENDABLE

Fig. 2-28 Planning Guidelines

LMSC-A990556

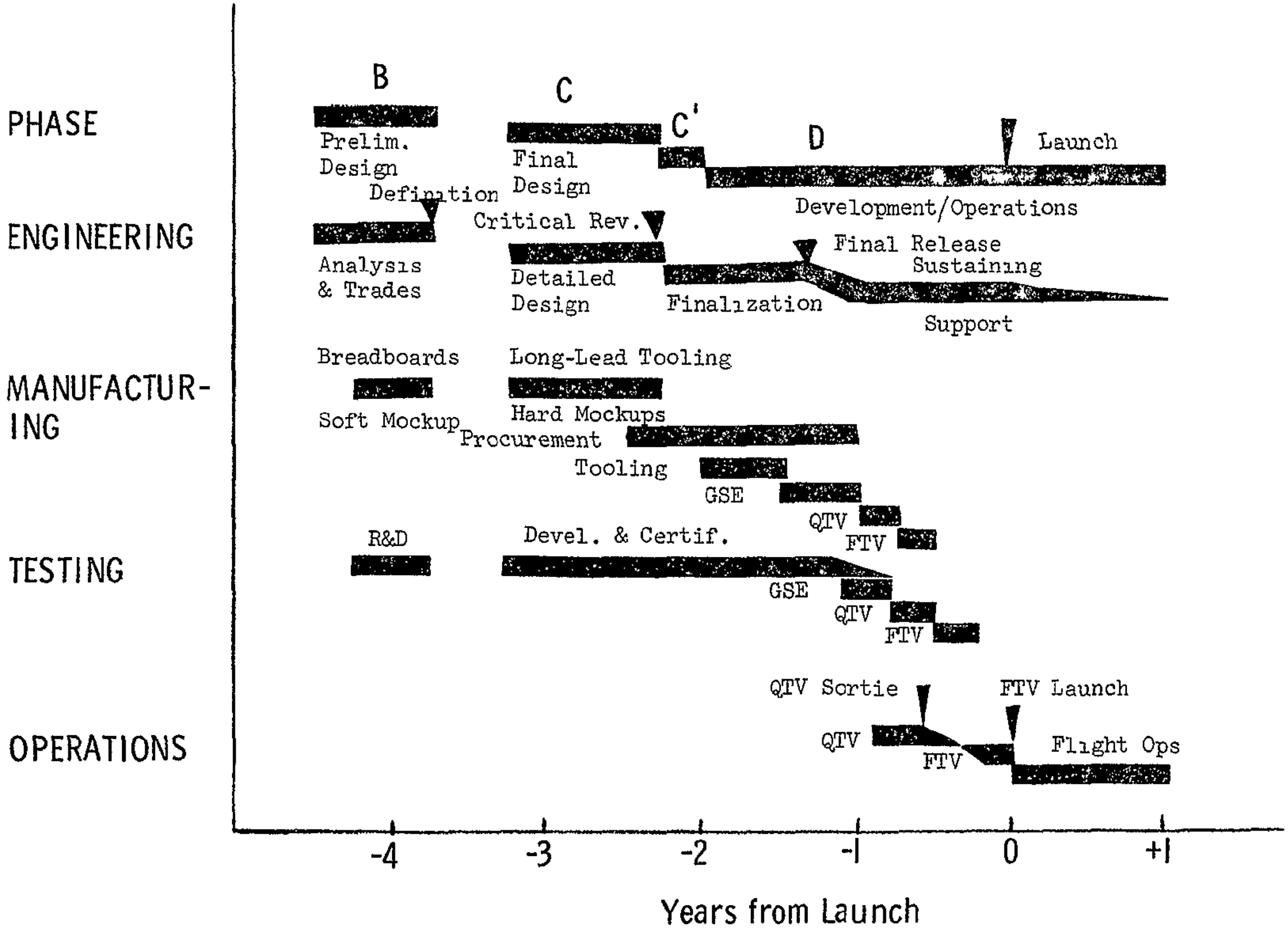
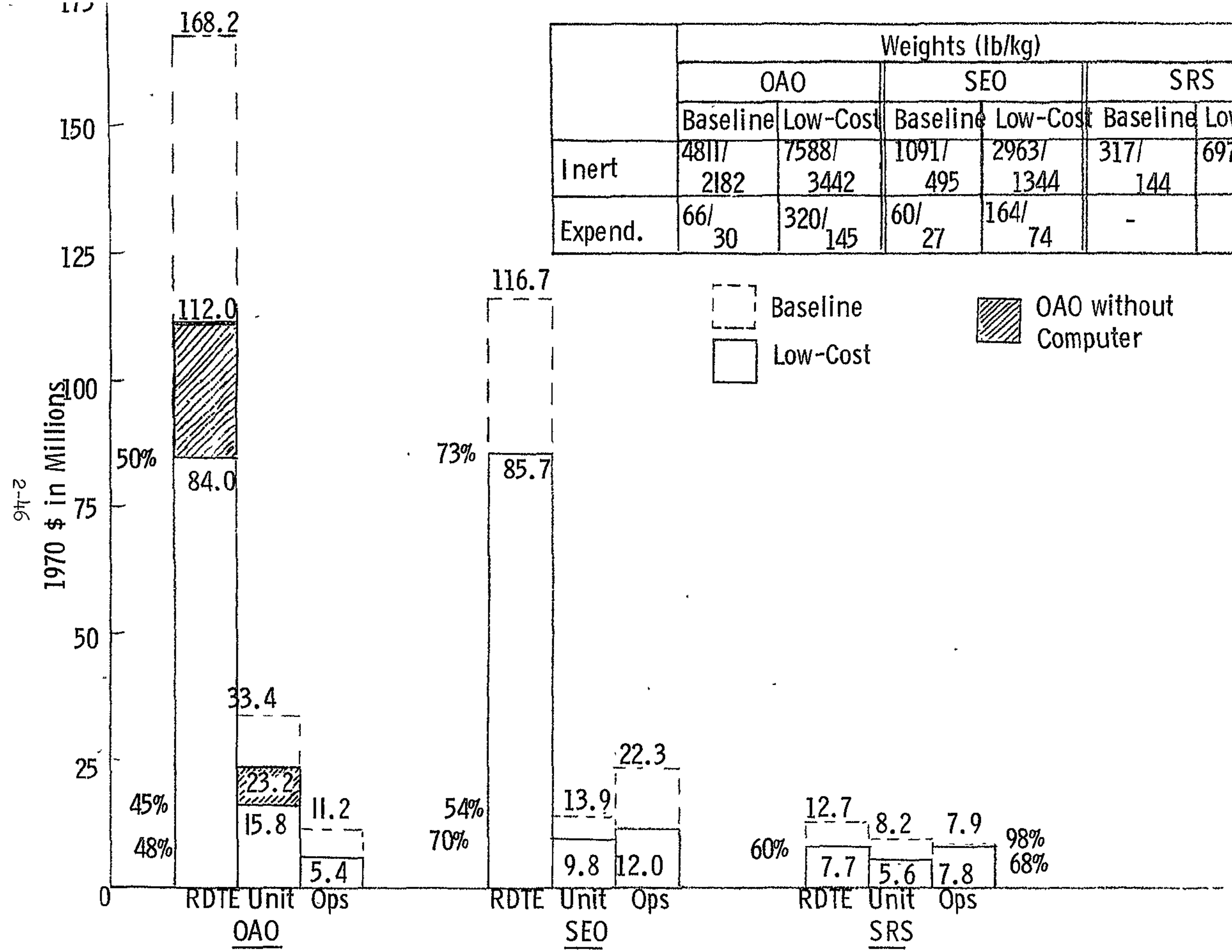


Fig. 2-29 Low Cost OAO-B Master Schedule (Space Shuttle-Launched)

- Bottom-up costing - using 1970 rates - includes labor, overhead, and G&A (no prime contractor fee)
- Engineering, Manufacturing, Test, and Operations cost estimates based on Program Plan
- Typical Program Management and Quality Assurance percentages applied
- Allowances included for:
  - Rework and scrappage
  - Engineering changes
  - Spares and Logistic Support
  - Tooling, GSE, and STE Maintenance
  - Computer Hours
- All cost spreads by subsystem and by year

Fig. 2-30 Estimating Approach for Low-Cost OAO and SEO



	Weights (lb/kg)					
	OAO		SEO		SRS	
	Baseline	Low-Cost	Baseline	Low-Cost	Baseline	Low-Cost
Inert	4811/ 2182	7588/ 3442	1091/ 495	2963/ 1344	317/ 144	697/ 316
Expend.	66/ 30	320/ 145	60/ 27	164/ 74	-	-

Fig. 2-31 Costs and Weights - Low-Cost Designs (Shuttle-Launched)

historical baseline, it was requested by NASA/HQ that LMSC estimate the costs of the baseline OAO and SEO payload programs: (1) using the same estimating methods employed on the low-cost payloads; but (2) using all the program approach and hardware of the baseline programs.

Recosting of the baseline programs was accomplished. The results are summarized in Fig. 2-32 (OAO-B) and Fig. 2-33 (SEO). There are some significant differences at the subsystem level but the total program costs are very nearly the same. It was thereafter assumed that the LMSC estimating methodology employed sufficient realism and conservatism so that the low-cost payload estimates could be used without multiplying by a "growth" factor.

2.5.4 Technology vs Payload Effects

As mentioned early in this report, a baseline requirement of the study was to use 1970 technology where possible to obtain a cost reduction. In general, technology did not influence the low-cost design approach in a cost-significant manner. The principal exception was in the OAO, where a 1970 state-of-the-art general-purpose computer was substituted for a fairly large quantity of electronic assemblies in the Stabilization & Control and CDPI subsystems. This type of fairly low-cost, reliable computer was not available for spacecraft application in the early 1960's (during the OAO development). The use of this computer accounted for a large percentage of the RDT&E and unit cost savings in the low-cost OAO. A separate cost-allocation analysis was performed, including recosting of the low-cost OAO without the computer substitution. The resulting costs indicated that the computer substitution accounted for about 35 percent of the total program savings on the OAO (33.6 percent of the RDT&E savings; 39.5 percent of the unit savings). The "without-computer" low-cost OAO provided about a 33 percent reduction in baseline RDT&E and unit cost when compared with the "with-computer" reduction of 50 percent. Figure 2-34 illustrates graphically the major contributors to the 50 percent cost reduction (equivalent to the 100 percent "savings" shown).

Costs in Millions $ / Subsystems	HISTORICAL COSTS				RECOSTED OAO-B			
	RDT&E Cost	Unit Cost	Unit Ops.	Total Prog.	RDT&E Cost	Unit Cost	Unit Ops.	Total Prog.
Adapter	$ 0.600	$ 0.150	$ 0.100	$ 0.850	$ 1.088	$ 0.166	$ 0.081	$ 1.335
Experiments	8.717	7.800	2.200	18.717	15.757	3.573	2.127	21.457
Structures & Mech.	9.044	5.100	0.020	14.164	11.020	1.156	0.393	12.569
Electrical	17.083	2.900	0.550	20.533	17.396	3.583	0.726	21.705
Stabilization & Cont.	78.469	11.700	3.900	94.069	72.292	14.302	3.599	90.193
Attitude Control	3.275	0.300	0.200	3.775	4.877	1.074	0.209	6.160
Communications, Data Processing, & Instrumentation	40.823	4.600	2.800	48.223	38.895	6.921	3.086	48.902
Environmental Control	6.045	1.000	0.550	7.595	5.028	0.973	0.298	6.299
Unallocated	1.353	2.600	0.900	4.853	1.311	0.217	0.473	2.001
TOTAL PAYLOAD	$165.409*	$ 36.150*	$ 11.220*	$212.779	$ 167.664	$ 31.965	$ 10.992	$ 210.621

*Initial baseline data from NASA/Goddard

Fig. 2-32 Comparison of Historical & Recosted OAO-B Costs (1970 $)

COST IN MILLION / SUBSYSTEM	ORIGINAL BASELINE (2 YR.)				RECOSTED BASELINE			
	RDT&E	Avg. Unit	4 Unit Ops.	Total* Program	RDT&E	Avg. Unit	4 Unit Ops.	Total* Program
Adapter	2.9	0.2	1.7	5.7	1.2	0.2	0.1	2.3
Experiments	47.1	2.5	9.3	69.1	46.9	3.0	5.6	67.6
Structures & Mechanisms	6.9	1.2	0.8	13.6	10.6	1.3	1.0	17.9
Electrical	12.4	1.9	1.9	23.9	14.1	2.1	2.1	26.4
Stabiliz. & Control	15.9	3.3	2.5	34.8	17.1	2.5	2.7	32.3
Att. Control	4.0	0.6	0.5	7.8	4.1	0.6	0.6	7.4
CDP & I	25.9	3.9	4.3	49.6	25.4	3.3	3.9	46.0
ECS	0.5	0.1	0.1	0.7	1.1	0.1	0.2	2.1
Unallocated	1.1	0.2	1.2	3.3	0.3	0.1	0.3	1.1
	116.7	13.9	22.3	208.5	120.8	13.2	16.5	203.1

* Includes 5 units

Fig. 2-33 Comparison of Historical & Recosted SEO Baseline Costs

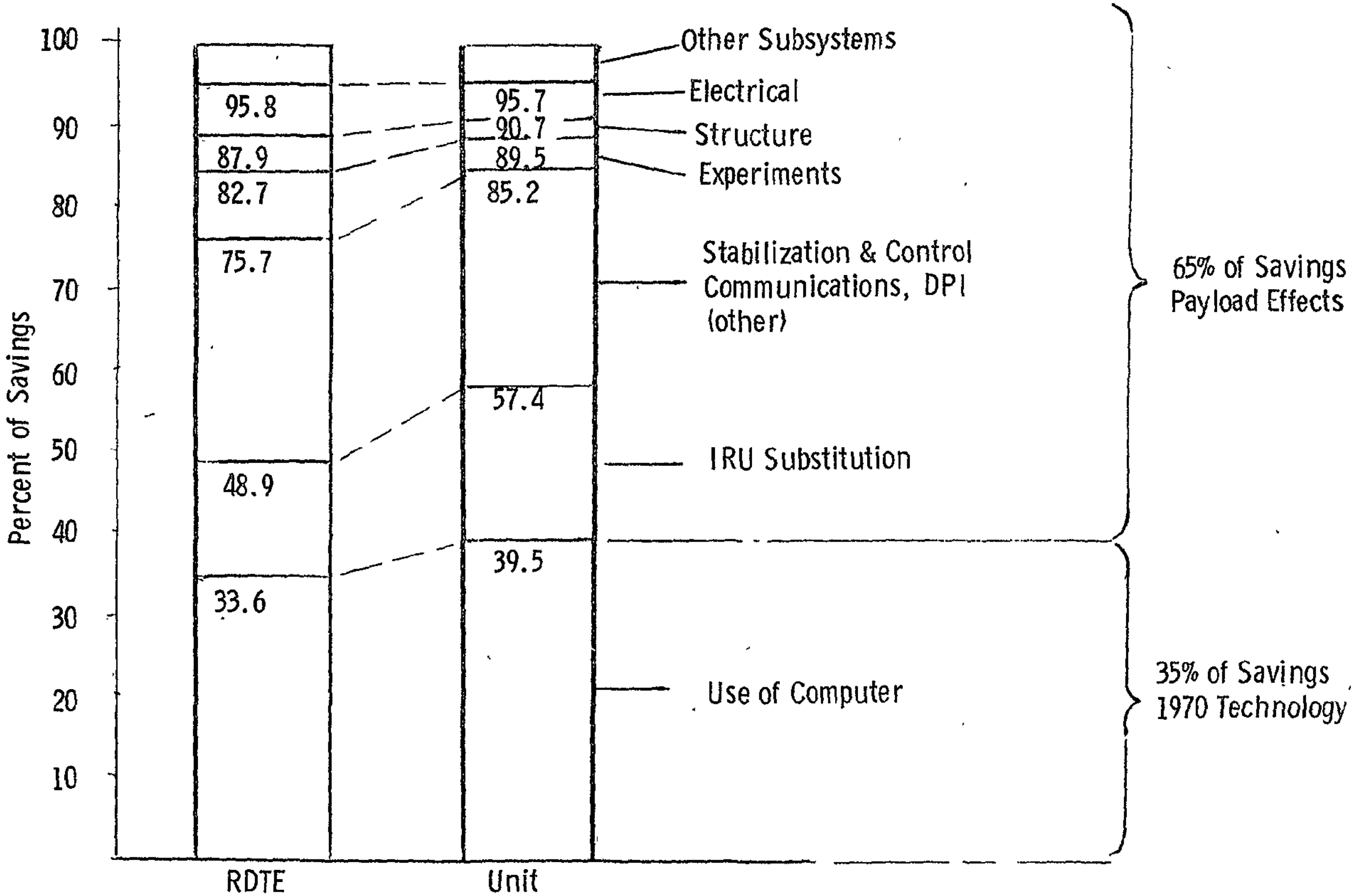


Fig. 2-34 Technology vs Payload Effects - OAO-B

2.6 IMPACT OF LOW-COST PAYLOADS ON TRANSPORTATION SYSTEM AND PROGRAM COSTS

Because of the strong dependence of the Shuttle-launched low-cost payloads upon the implementation of payload-compatible interfaces, it was determined necessary to verify that the LMSC-proposed interfaces with the Shuttle system were feasible and practicable. Conceptual designs were therefore created for: (1) a payload deployment/retrieval gear and, (2) a payload checkout set for on-orbit use with the Shuttle. Also, a complete concept for repair and refurbishment was developed for payloads on orbit and modules and components on the ground.

These concepts are discussed in detail in Section 8 of the report. A summary of approaches and results is presented following.

2.6.1 Payload/Shuttle Interfaces

In obtaining the maximum cost benefit from the low-cost payloads, it seemed desirable to adapt the payloads and the supporting Shuttle systems to "a launch base on orbit". In this manner, the failures experienced in launch/ascent could be repaired on orbit prior to payload deployment from the Shuttle. This concept required: (1) the use of on-orbit checkout by Shuttle-carried payload checkout equipment and, (2) the design of payloads to allow easy repair, refurbishment, and reuse, it was necessary to provide an installation which could be employed for these operations with various payloads. The elements of the payload support equipment and interfaces are listed on Fig. 2-35 with the low-cost OAO extended on deployment booms. These interfaces have been investigated, preliminary requirements established, and payload compatibility with the Shuttle has been verified.

2.6.2 Payload Support, Deployment, and Retrieval

A universal-usage deployment/retrieval gear was conceptually designed. The principal hardware elements are bi-stem extendable booms, smaller sizes of

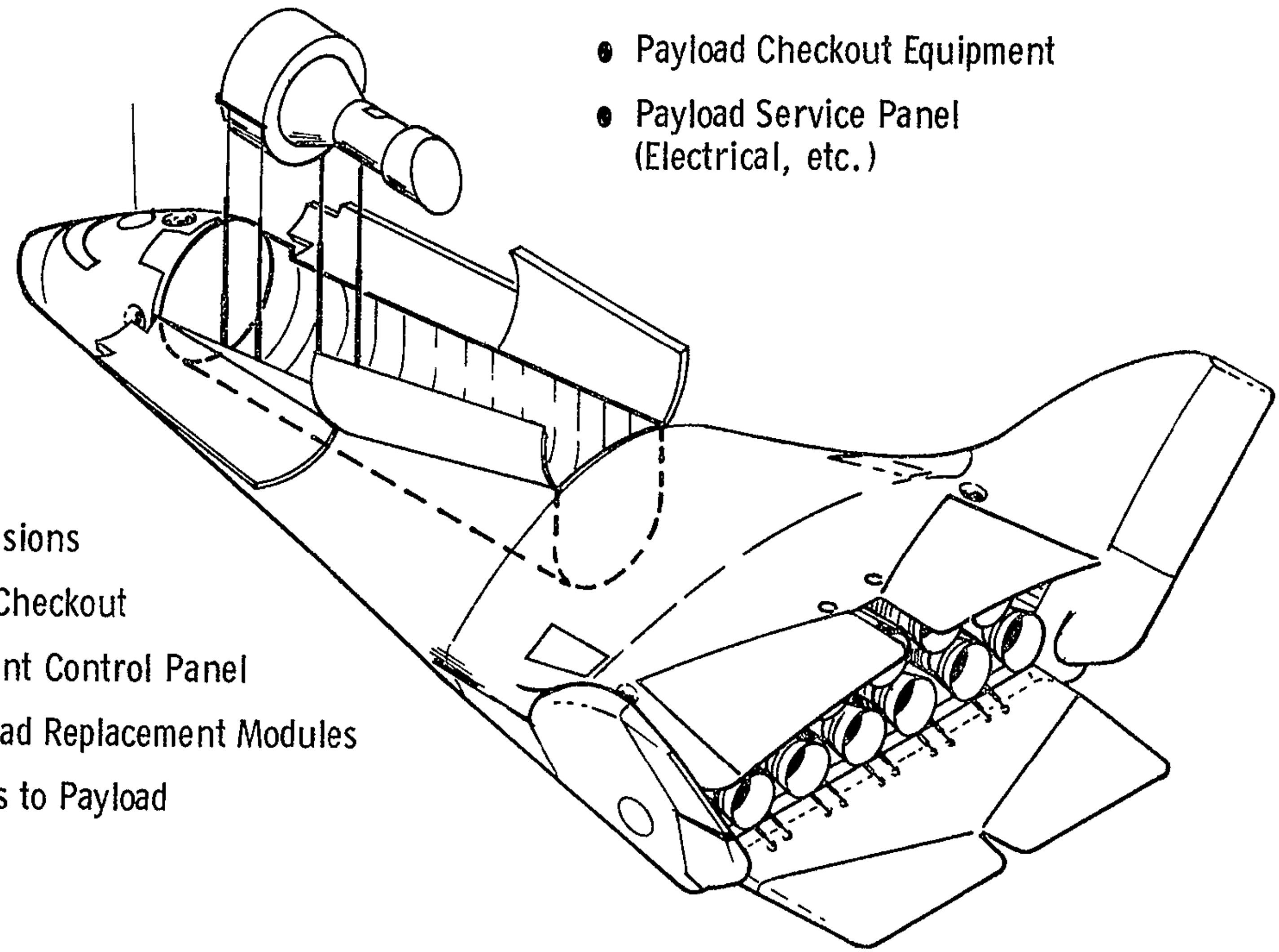


Fig. 2-35 Shuttle Interfaces with Low-Cost Payloads

which have been successfully used on previous spacecraft applications. These booms, installed as a single unit, in pairs, or as a set of four, can extend or retract a payload from its base mounting position in the Shuttle cargo bay.

A scale drawing of the installation of the SEO/Tug in the Shuttle is shown on Fig. 2-36. Six latching hold-down supports (1L/1R on the SEO and 2L/2R on the Tug) sustain all launch/ascent, maneuvering, reentry, and landing loads. The booms operating in zero-g are stiff enough to sustain the bending loads applied by minor maneuvering of the Shuttle even with booms extended. The booms can sustain reasonable loads even in 1-g load field and can be readily tested on the ground with simulated weights attached. Figure 2-37 shows the SEO/Tug extended on the booms and a support cradle assembly which comprises remote-actuated latches (energized via electrical cable which is reeled out within each boom) at each of the four corners. Retrieval of the SEO/Tug is accomplished by engagement of the four support pins on the Tug into the mating drogue funnels on the extended cradle assembly. Positioning for engagement can be accomplished by vernier control of the Tug, by use of telefactor robot, by separate "grappling" mechanism deployed from the Shuttle, and/or by use of crew in EVA with strap-on thruster devices.

2.6.3 On-Orbit and Standardized Checkout

On-orbit checkout of payloads provide specific advantages:

- Greatly increases probability of successful mission by allowing elimination of launch/ascent failure contribution
- Allows lower payload design reliability and concomitant reduced cost
- Allows payload - cognizant personnel to perform first-hand observation of payload operating in orbit environment
- Makes feasible on-orbit module replacement and ré-checkout for repair or refurbishment

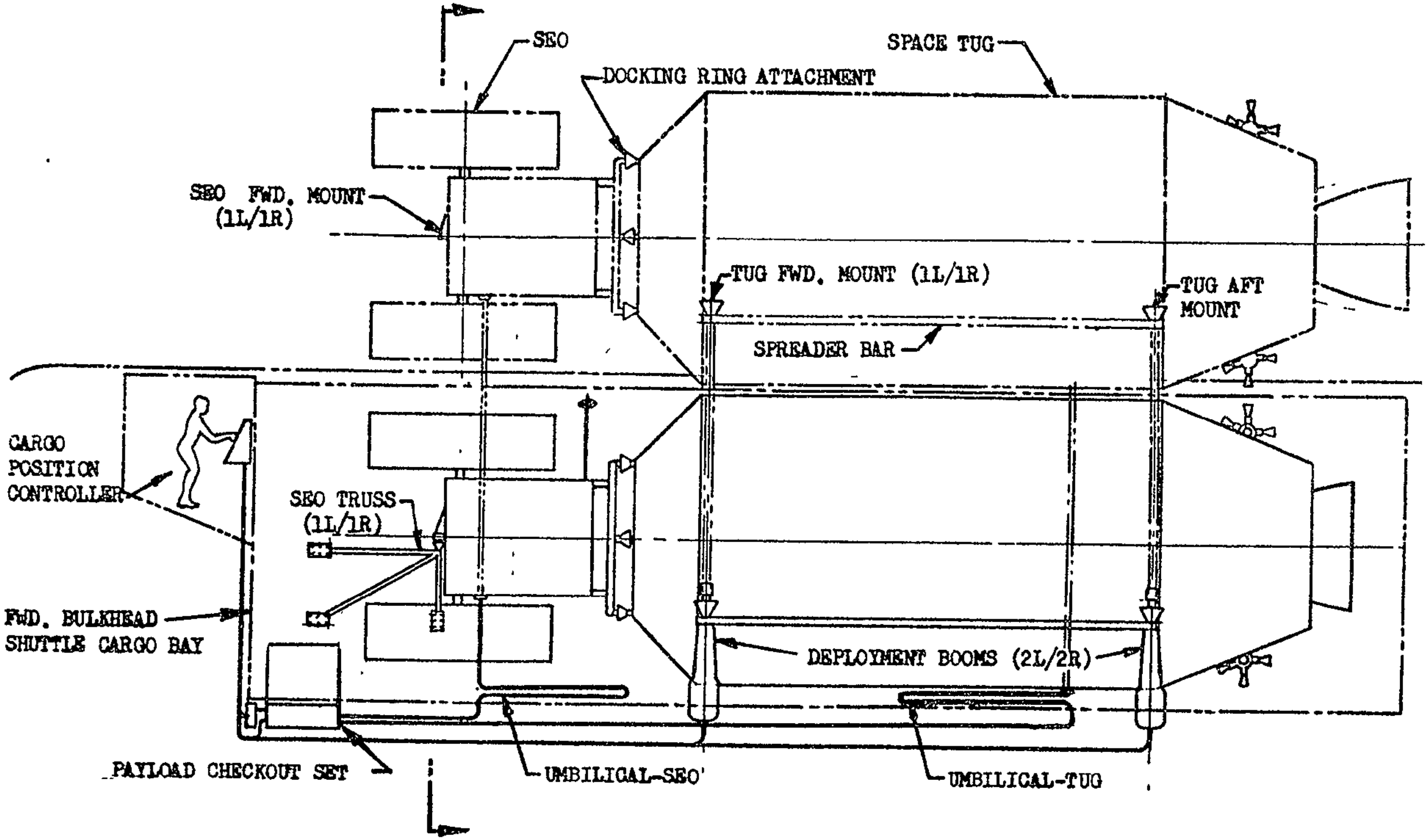


Fig. 2-36 Shuttle Installation of SEO and Space Tug

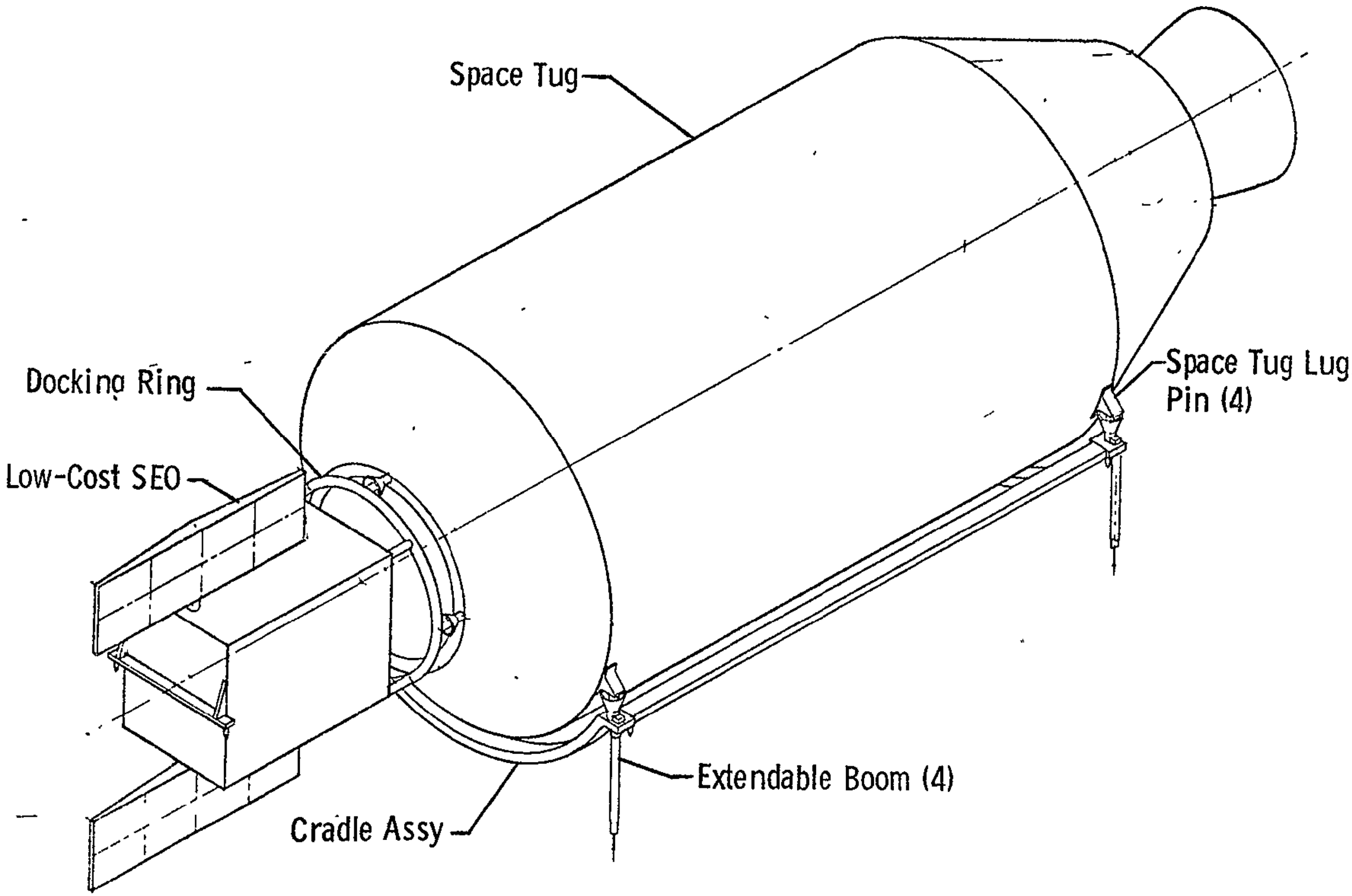


Fig. 2-37 SEO/Tug Extended Position for Deployment/Retrieval

A phased-checkout approach was developed so that payloads could be exposed to a series of verifications, using the same checkout set. The seven phases of test/checkout proposed are shown on Fig. 2-38. Phase I is accomplished at the payload production plant and Phase II is accomplished at the launch base prior to mating of the payload into the Shuttle. Phases III through VII are conducted with payload mounted in the Shuttle and using a Shuttle-carried payload checkout set. Considerable analysis of this concept has been done and detail checkout lists have been created for the OAO, equivalent to the historical ground checkout requirements for the OAO-3 payload (reference data supplied by NASA/GSFC). It has been determined that the concept is feasible and desirable. Further, actual concept design of a Shuttle-carried checkout set has been developed and weight, volume, and cost estimates made.

An extension of the payload checkout set, standardized checkout equipment, has also been investigated and also determined to be feasible. The qualitative cost reduction aspects of this concept are listed on Fig. 2-39.

2.6.4 Repair, Refurbishment, and Reuse of Payloads

The single most important cost driver in the unmanned payload cost-reduction effort is the repair, refurbishment, and reuse of payloads. A methodology was therefore developed to validate the feasibility and quantitative cost data were derived for repair/refurbishment which could be used for application to the total mission model by Aerospace Corporation and Mathematica. A complete description is included in sub-section 8.4 of this report. A brief resume is provided following.

2.6.4.1 Investigation of Hardware and Operational Factors. A number of factors were considered and analyzed during establishment of the proposed payload repair/refurbishment approaches; they are listed on Fig. 2-40. Some of the possible modes of in-orbit repair/refurbishment are conceptually pictured in Fig. 2-41. Although the pressurized IVA (shirt sleeve) mode was initially desired by NASA in the early Shuttle design phase, it has been removed as a mandatory requirement in favor of EVA or non-pressurized IVA modes for crew direct

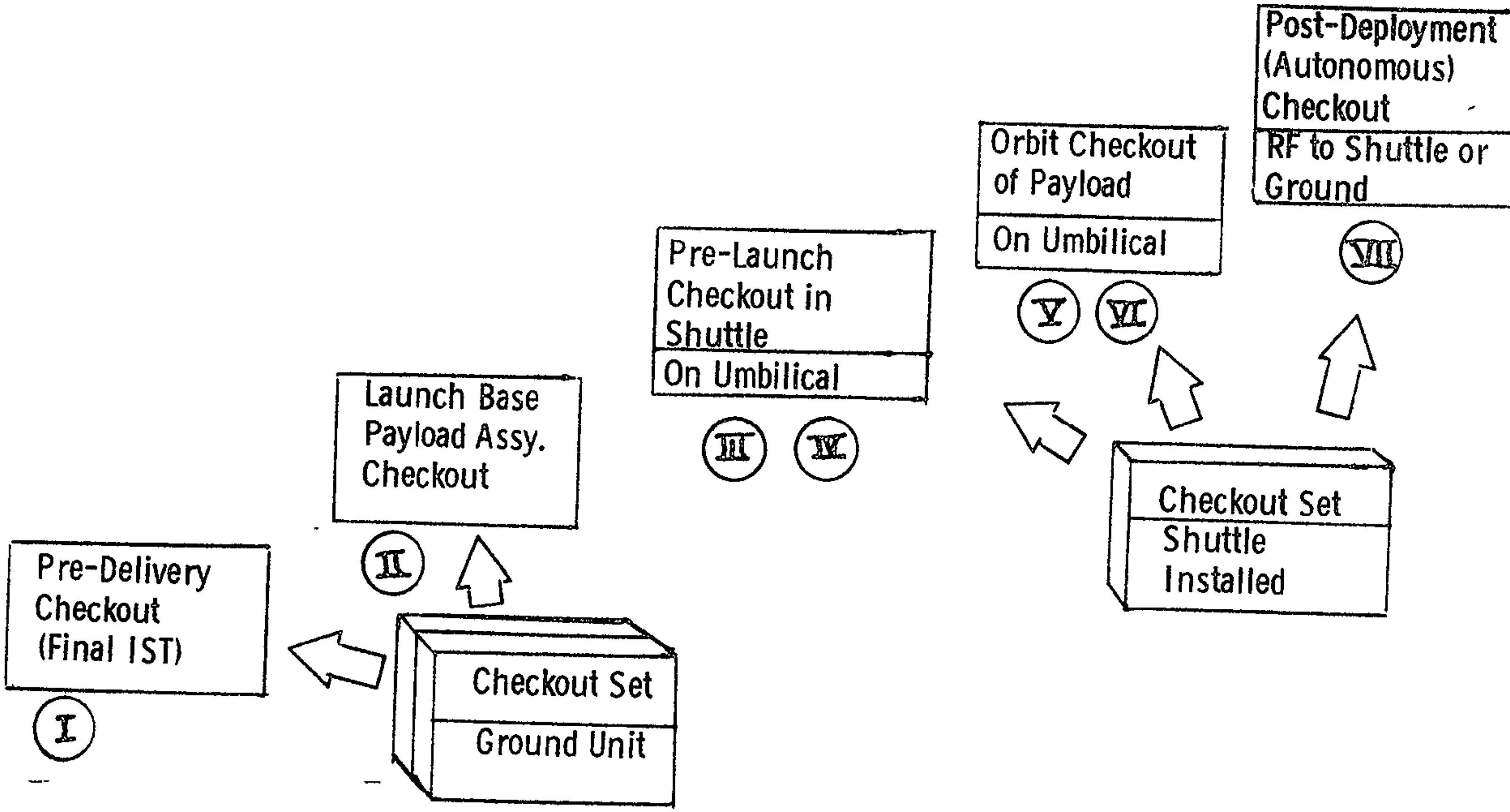


Fig. 2-38 Phased Test & Checkout Approach - Low-Cost Payloads with Shuttle

- USE OF "STANDARD" SUBSYSTEM APPROACH ALLOWS STRONG CONSIDERATION OF MATCHING STANDARD CHECKOUT EQUIPMENT
- ALLOWS STANDARDIZATION OF FAULT ISOLATION AND LOWER DIAGNOSIS COSTS
- A STANDARD CHECKOUT SET APPROACH WOULD PERMIT REDUCTION IN RDT&E COSTS WHEN COMPARED TO THE SEPARATE DEVELOPMENT FOR EACH MISSION OF A PAYLOAD-PECULIAR CHECKOUT SET
- PERMITS REDUCTION OF RECURRING COSTS BY LARGE-QUANTITY PRODUCTION OF STANDARD CHECKOUT SETS TO BE USED WITH ALL PAYLOAD SUBSYSTEMS (COULD BE GFE)
- ALLOWS COROLLARY STANDARDIZATION AND COST REDUCTION OF SHUTTLE MATING INTERFACES
- ALLOWS GENERALIZED TRAINING OF TEST/CHECKOUT CREWS

Fig. 2-39 Cost Reductions with Standardized Checkout Equipment

- DEACTIVATION AND ATTITUDE STABILIZATION OF PAYLOAD
- EVA VS REMOTE MANIPULATORS
- MANNED AND UNMANNED SPACE TUG OPERATIONS WITH THE SHUTTLE
- RECALIBRATION OF PAYLOAD AFTER MODULE REPLACEMENT
- GROUND-BASE RESPONSE TIME
- TURN AROUND TIME FOR GROUND REFURBISHMENT
- ON-ORBIT RE-CHECKOUT AFTER MAINTENANCE/REFURBISHMENT
- PAYLOAD WEAROUT/RELIABILITY VS REFURBISHMENT CYCLE

Fig. 2-40 In-Orbit Maintenance/Refurbishment Analyses

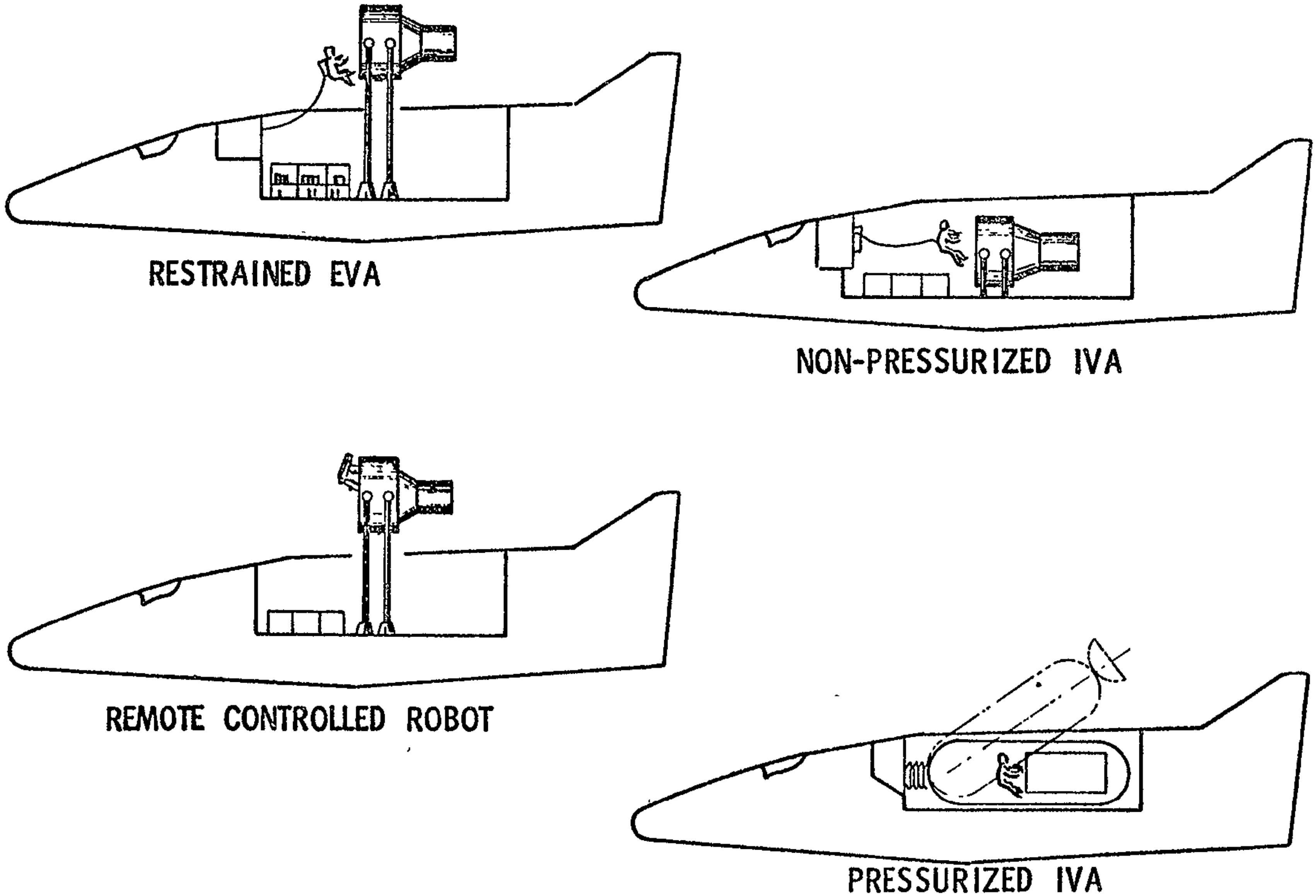


Fig. 2-41 In-Orbit Maintenance/Refurbishment Concepts

access to payloads in combination with automated payload handling devices. An automated module replacement device could be readily substituted for the robot mode pictured; NASA/GSFC has done preliminary work on such a device for use with the Large Stellar Telescope.

2.6.4.2 Basic Approach to Repair/Refurbishment. Figure 2-42 provides a basic tabulation of the four elements involved in repair and refurbishment of the OAO and SEO low-cost payloads.

2.6.4.3 Cost Savings with Payload Refurbishment. The savings attainable with the proposed refurbishment at the payload, module, and component level are extremely significant. Figure 2-43 is a tabulation of results obtained from the OAO and SEO refurbishment analyses. At the payload system level, the refurbishment approach provides a 39 percent saving for an OAO 6-year program and a 41 percent saving for a 10-year SEO program (compared with a low-cost expendable-launched payload). When the launch and operations costs are made part of the total, the savings increase to 50 percent for both the OAO and SEO programs.

2.6.4.4 Refurbishment Cost Ratios. Summary calculations were made of the cost of a refurbished OAO or SEO. The "average" refurbished OAO would cost 32.5 percent of the unit recurring cost of a new OAO, with refurbishment performed on a one-year time cycle. The average refurbished SEO would cost 39 percent of a new SEO, with refurbishment performed on a two-year time cycle.

2.7 STANDARD SPACECRAFT AND SUBSYSTEMS

As a separate task of the study (Task 3), not directly contributing to the cost reductions documented for the low-cost OAO, SEO, and SRS payloads; the concept of a standard spacecraft was investigated to: (1) ascertain the technical feasibility and, (2) determine the economic desirability. This effort is described in detail in Section 7 of this report; a highlight summary is provided following.

* REPAIR ON ORBIT
  * Carry as spares on initial launch 10 different modules (total wt. 2093 lb) - OAO
    11 different modules (total wt. 1423 lb) - SEO
  * Checkout of payload on orbit
  * Replace any module which has failed or degraded in launch/ascent
  * Return failed module for ground refurbishment
* REFURBISHMENT OF PAYLOAD
  * Periodically replace the orbiting payload with a refurbished (at nom. 1-yr. interval for OAO; at 2-yr. intervals for SEO)
  * Retrieve the "used" payload from orbit with Shuttle or Tug/Shuttle and return to earth.
  * Remove used/failed modules from space frame
  * Install new (or refurbished) modules
  * Perform system-level payload checkout
* REFURBISHMENT OF MODULES
  * Remove module cover and equipment components
  * Install new (or refurbished) components into module
  * Test module in spacecraft simulator, using standard checkout set
* REFURBISHMENT OF COMPONENTS

Fig. 2-42 Low-Cost Payload Repair/Refurbishment Approach

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COST ELEMENT	OAO (6-Year Program)*		SEO (10-Year Program)**	
	Expendable-Launched	Shuttle-Launched	Expendable-Launched	Shuttle-Launched
Non-Recurring	$ 89.41 M	$ 84.03 M	$ 97.99 M	$ 85.70 M
Unit Cost - Delivered Payloads	114.84	15.81	234.56	49.15
Average Refurb. Ratio	-	.325/1 yr.	-	.390/2 yrs.
Payload Module/Component Refurb	-	25.29	-	60.88
Payload Totals	204.25	125.13	332.55	195.73
Launch Costs	108.00	18.00	318.00	102.00
Operations Cost	40.02	31.05	67.80	59.75
Total Program Cost	$ 352.27 M	$ 174.18 M	$ 718.35 M	$ 357.48 M
Program Savings →	$ 178.1 M		$ 360.9 M	

* Refurbishment performed at 1-yr. cycle intervals
** Refurbishment performed at 2-yr. cycle intervals

Fig. 2-43 Savings with Payload Refurbishment

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2.7.1 Feasibility of a Standard Spacecraft

Early study of the basic concept of a standard spacecraft indicated that it was technically feasible. In analysis of the NASA Mission Model, a large percentage (86 percent) of payload programs were potentially suitable for some combination of experiments onto one or more spacecraft (see Fig. 2-44). Further analysis revealed that unmanned missions also could be grouped by orbit and general scientific objective. Two examples of this are shown in Fig. 2-45 for low-altitude/28.5° orbits and for polar/sun-synchronous orbits. Finally a preliminary analysis of the spacecraft support for the various candidate missions revealed that a single spacecraft with quite broad subsystem capability could accommodate the majority of all missions. The characteristics of this hypothetical standard spacecraft are listed in Fig. 2-46. Combining these preliminary conclusions with data from previous LMSC effort on design and manufacture of standard spacecraft elements, it was determined that the standard spacecraft was indeed technically feasible.

2.7.2 Basic Approach to Design of Standard Spacecraft Hardware

Basic conclusions, involving preliminary economic considerations, and relevant to the approach to standard spacecraft were developed early.

- Rather than a single all-coverage spacecraft to accommodate all missions, there may be a small group of standard spacecraft

- Even though a standard spacecraft were developed, there may be certain space missions which can be more economically supported by specialized designs

- The real foundation for a standard spacecraft appears to be the development of standard subsystems and/or modules thereof

A HIGH PROPORTION OF FUTURE UNMANNED PAYLOAD PROGRAMS ARE POTENTIALLY SUITABLE FOR SPACECRAFT SHARING:

	Qty. of S/C	% of Traffic Model
PROGRAMS DEFINITELY UNSUITABLE FOR COMBINATION	69	14%
PROGRAMS POTENTIALLY SUITABLE FOR COMBINATION		
- Low Altitude, East Launch ETR	59	12
- Polar - Sun Synchronous		
ERTS	72	14
METSAT	16	3
- Synchronous Equatorial		
ERTS	57	11
Comm/Nav	148	30
Highly Eccentric	80	16
	432	86%

Fig. 2-44 Feasibility of Standard Spacecraft - Multiple Experiments

LOW ALTITUDE - 28.5^{0}

Mission	Alt. (N.M.)	Exper. Weight (lb)	Qty. S/C	Exper. Power	Point. Accur.	Start Year	Program Duration
Large Stellar Telescope	350	8270	1	1.5 KW	10 sec	1980	11 yr.
Large Stellar Observatory	350	7520	1	1.0 KW	1 deg	1982	9 yr.
HEAO	230	12500	2	265 W	5 sec	1979	12 yr.
Large Radio Observatory	350	10000	1	2.0 KW	10 sec	1984	7 yr.
Astronomy Explorers	270	250	15	50 W	3 deg	1978	13 yr.

POLAR - SUN SYNCHRONOUS

Mission	Alt. (N.M.)	Exper. Weight(lb)	Qty. S/C	Exper. Power	Point. Accur.	Start Year	Program Duration
Polar EOS	500	870	12	400 W	4 min	1980	13 yr.
Earth Physics	400	150	7	150 W	10 sec	1980	13 yr.
Polar ERS	500	850	28	400 W	4 min	1978	13 yr.
TIROS	700	245	14	100 W	3 deg	1978	13 yr.

Fig. 2-45 Common Low-Earth Orbits and Spacecraft Requirements

STANDARD SPACECRAFT WITH SUBSYSTEM CAPABILITY TO MEET FOLLOWING REQUIREMENTS WILL ACCOMMODATE THE MAJORITY OF ALL MISSIONS:

GNC	Earth or inertial orientation $1/4^{\circ}$ attitude reference with option for 10 arc sec Stabilization by reaction wheels and/or mass expulsion
TT&C	Wide-band data link with up to 1 megabit capability Choice of transmission powers up to 50 watts Data recording capability up to 1 MHz
Electrical	Modular capability from 350 to 1050 watts average power, $\pm$ 28 VDC regulated or unregulated, 115 VAC, 1, 2 or 3-Phase
Environmental Control	Provided for standard spacecraft exclusive of experiment subsystem, for earth orbit or equivalent

Fig. 2-46 Standard Spacecraft for Unmanned Payloads

2.7.3 Standard Subsystems

The basic concept of standard subsystems was developed by listing separately all of the mission requirements by spacecraft subsystem and then consolidating these requirements into a minimum quantity of variants. The subsystem characteristics for these variants, some of which can be built up by using multiples of a single module (such as for the electrical subsystem), are shown on Fig. 2-47. These variants were applied to the NASA Mission Model and the total number of applications determined by quantity of missions and quantity of spacecraft required per mission. A sample sheet of this application analysis is shown as Fig. 2-48. These data were transferred to the economic analysis totals for standard subsystems.

2.7.4 Standard Spacecraft

A typical set of subsystem variants were selected which could accommodate a reasonably large quantity of missions. Subsystem modules were concepted and developed into a set of "standard" modules; one of these is shown in Fig. 2-49. These modules were then arranged into an overall spacecraft configuration which is illustrated on Fig. 2-50. Each module, as with the previously-described low-cost payloads, is readily replaceable in orbit by a Shuttle crew member or by automated module handling devices. The standard spacecraft design effort was accomplished to the depth required to verify that the concepts were feasible and could be implemented in an actual hardware program. The eventual standard spacecraft may be different in actual configuration from the typical one developed; however, the basic characteristics to provide compatibility with the Shuttle system and to allow on-orbit checkout, repair, refurbishment, and equipment and experiment update must be maintained.

2.7.5 Potential Savings with Standard Spacecraft Hardware

Two economic analyses were performed: (1) determination of the savings accruing from the use of standard subsystems applied throughout the NASA Mission Model, and (2) determination of overall savings resulting from the use of a

SUB-SYSTEM TYPE	ELECTRICAL (BATTERY & SOLAR ARRAY)	GUIDANCE, NAVIG., STABILIZATION, CONTROL	TELEMETRY (S/C DATA & COMMAND)	COMMUNICATION-(EXPERIMENT DATA) S-BAND*
A	• 350 W • Sun-Orient. Solar Array	• Stellar/Solar Ref. • 10 arc sec • Inertial Platform	• 8 - 33 BPS • 50 W XMTR (IP) • Omni. Ant.	• 10^3 - 10^4 BPS • 50W XMTR (IP) • 10-30 Hi-Gain Tracking Antenna
B	• 700 W • Sun. Or. S/A	• Stellar/Solar Ref. • 15 arc min. • Inertial Platform	• 10^4 BPS • 2 W XMTR (SEO) • Omni Antenna	• 2 x 10^6 BPS • 2 W XMTR (SEO) • 3 Ft. Fixed Hi-Gain Antenna
C	• 1050 W • Sun. Or. S/A	• Earth Reference • 15 arc min • Inertial Platform	• 10^5 BPS • 2W XMTR (LEO) • Omni Antenna	• 10^7 BPS • 2 W XMTR (SEO) • 6 ft. Hi-Gain Tracking Antenna
D	• 100 W • Body-Mounted Spinning Array	• Spin Stabilized • Axis Orientation Control		• 2 x 10^7 BPS (LEO) • 2W XMTR (Spin) • Toroidal Antenna
E		• Earth Ref. & Star Tracker • 4 arc min • Inertial Platform		• 10^6 BPS • 2 W XMTR (LEO) • Omni Antenna

* 30 ft. dia. ground receiver antenna, except for interplanetary which uses 210 ft antenna.

Fig. 2-47 Standard Subsystem Types & Characteristics

LMSC-A990556

MISSION		ORBIT		NO. OF SPACE-CRAFT	STANDARD SUBSYSTEM TYPE/QTY				
NAME	NO.	ALTITUDE	INCL.		GNSC	COMM	T/M	ELECT	REMARKS
			NASA EARTH OBSERVATION						
Polar EOS	21	500/500	SS	12	E 12	E 12	C 12	B 12	
Sync EOS	22	Sync	0°	6	E 6	C 6	B 6	B 6	
Earth Physics	23	400/400	90°	7	E 7	C 7	C 7	A 7	
Sync Met	24	Sync	0°	2	C 2	C 2	B 2	A 2	
Tiros	25	700/700	SS	3	C 3	B 3	C 3	A 3	
Polar ERS	26	500/500	SS	6	E 6	E 6	C 6	B 6	
Sync ERS	27	Sync	0°	7	E 7	C 7	B 7	B 7	
			NASA COMMUNICATION/NAVIGATION						
ATS	28	Sync	0°	7	E 7	C 7	B 7	--	8 kw
Small ATS Sync	29	Sync	0°	12	C 12	C 12	B 12	B 12	
Low	30	3000/300	90°	12	C 12	E 12	C 12	B 12	
Co-Op ATS Sync	31	Sync	0°	2	C 2	C 2	B 2	B 2	
Low	32	3000/300	90°	2	C 2	E 2	C 2	B 2	
Medical Net	33	Sync	0°	2	C 2		B 2	C 2	
Educ. Broadcast	34	Sync	0°	2	C 2	Comm. part of P/L	B 2	--	2 kw
Follow-on Sys.Demo.	35	Sync	0°	20	C 20		B 20	C 20	
Tracking/Data Relay	36	Sync	0°	10	C 10	C 10	B 10	B 10	
Planetary Relay	37	Sync	0°	9	C 9	C 9	B 9	A 9	

Fig. 2-48 Applicability of Subsystem Options to Missions

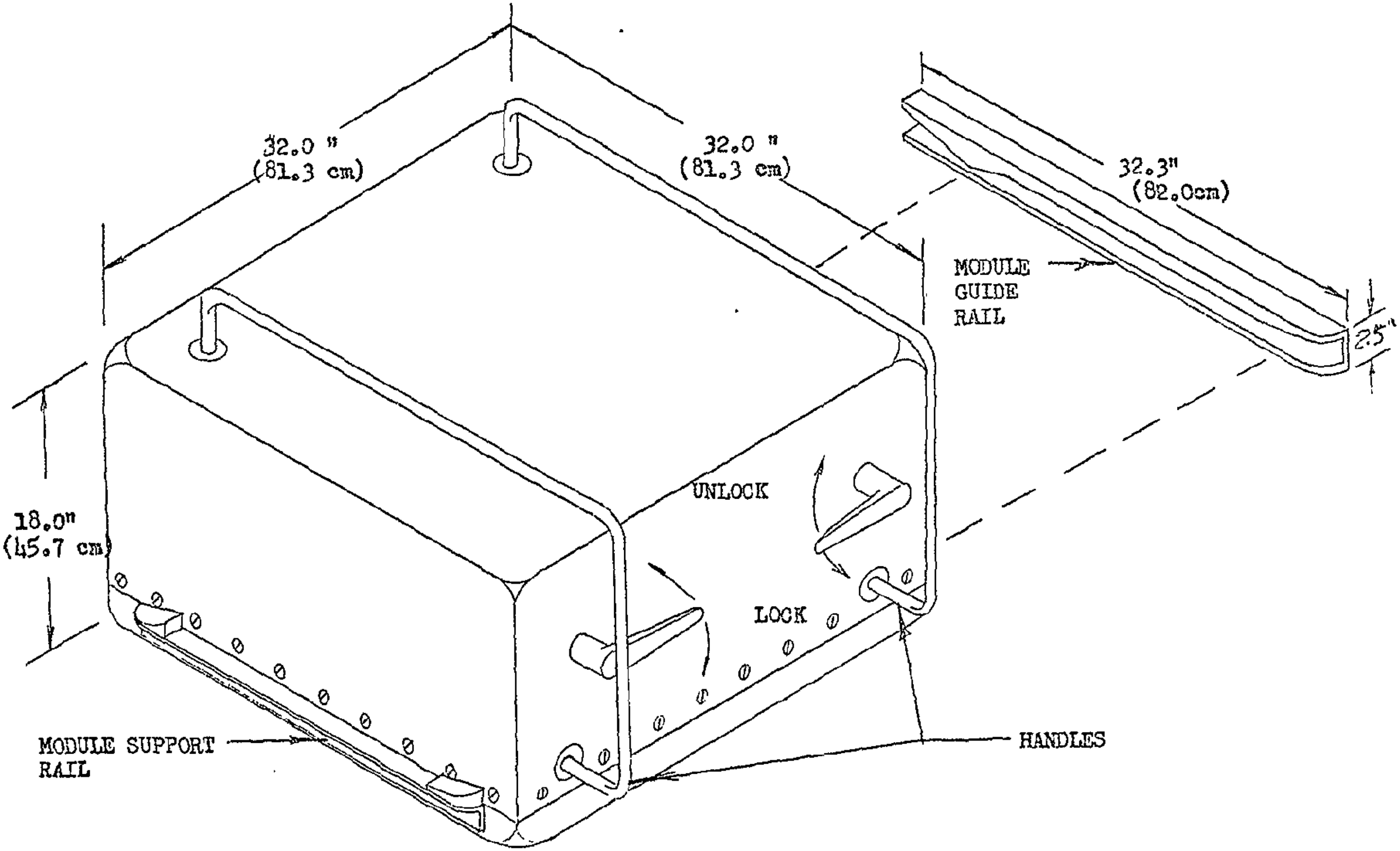


Fig. 2-49 Typical Standard Spacecraft Module

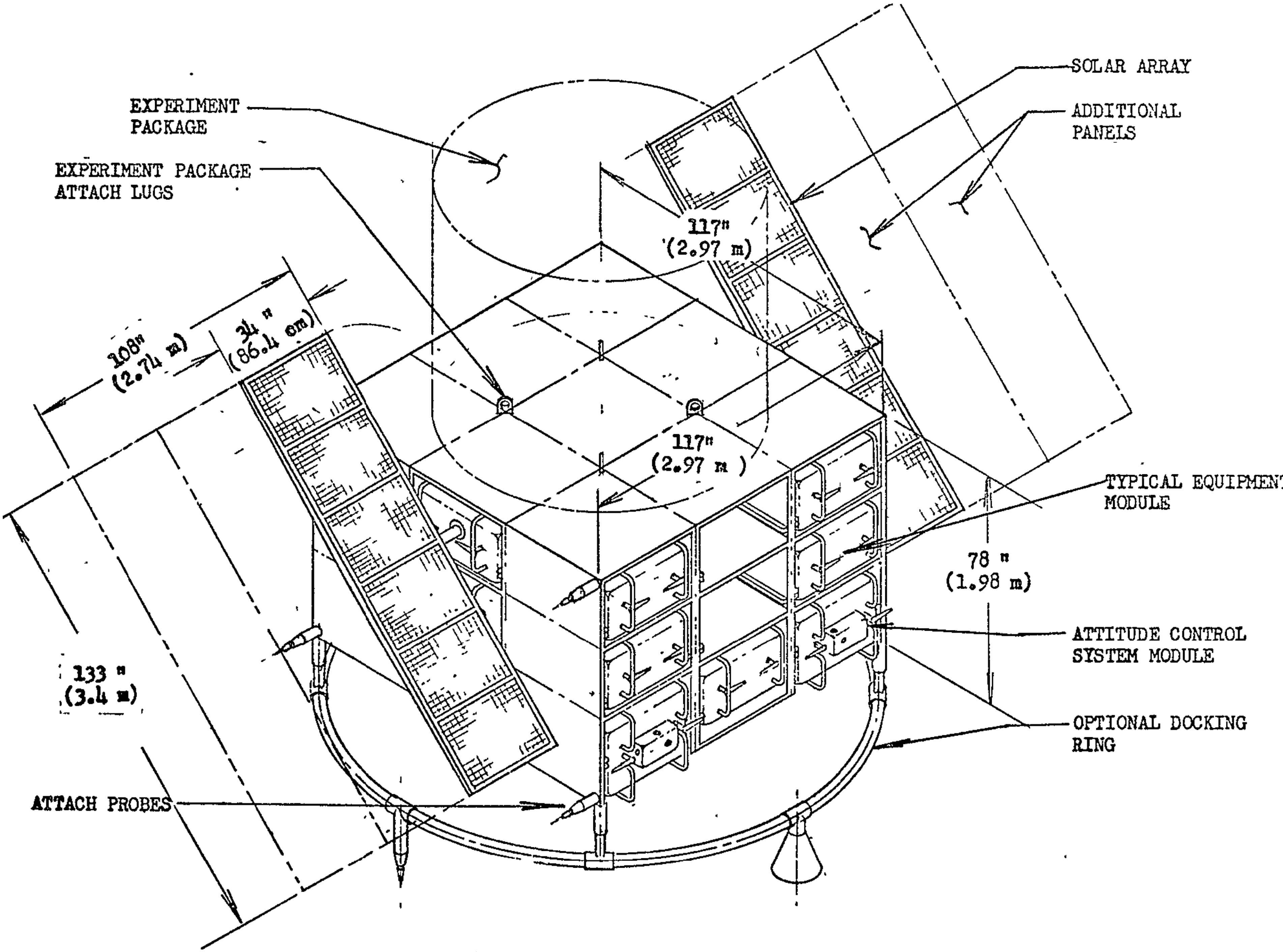


Fig. 2-50 Standard Spacecraft General Configuration

typical standard spacecraft (itself comprising standard subsystem elements) for a majority of the missions in the Mission Model. The primary cost saving of course is in the sharing of hardware development costs by a number of payload programs rather than having a separate project-peculiar development for each program.

The results of a typical sub-analysis on a standard electrical power subsystem applied to varying quantities of programs and spacecraft are tabulated on Fig. 2-51. Modular-design subsystems were used as the cost base. A 300-watt capacity subsystem could be applied to 17 payload programs comprising 52 spacecraft and result in a saving of $60.2 million. Increasing the subsystem capacity to 600 watts would "capture" 30 missions and 165 spacecraft and provide savings of $133 million. Using an approach where a modular subsystem could provide 300, 600, or 1200 watt capacity (modules selected as needed to fit the particular program requirement); the savings could be further increased to $231 million, covering 45 programs and 264 spacecraft.

The economic impact in RDT&E of applying standard subsystems to 53 missions of the Mission Model is shown in Fig. 2-52. Additional savings of $2262 million are indicated, as compared to the development of low-cost subsystems specially designed and developed for each of the 53 missions. This saving could be reduced somewhat by the higher average unit cost of the standard subsystems (because of capability "overkill" when compared to the project-peculiar subsystem.

In Fig. 2-53, there is shown the economic results of applying a single standard spacecraft to selected missions of the Mission Model. The cost savings accrue because a single-spacecraft development is substituted for the development of a large quantity of special spacecraft.

2.8 DESIGN GUIDELINES FOR FUTURE PAYLOAD DESIGN

Task 4 of the study was devoted to preparation of a Design Handbook for payload designers. The data developed will be issued as a separate contract end-item document and will not be discussed in detail in this final report.

SUBSYSTEM TYPE	MODULE CAPACITY (WATTS)	TRAFFIC CAPTURED		POTENTIAL SAVINGS ($ M)
		NO. OF PROGRAMS	NO. OF SPACECRAFT	
SINGLE MODULE	300	17	52	60.6
	600	30	165	133.0
MULTI-MODULE	600/1200	40	218	210.4
	300/600/1200	45	264	231.4

Fig. 2-51 Savings Potential - Standard Electrical Power Subsystem

SUBSYSTEM	OPTION	PROGRAMS	RDTE ($MILLION) PROGRAM PECULIAR PER PROGRAM	RDTE ($MILLION) PROGRAM PECULIAR CUM. TOTAL	RDTE ($MILLION) STANDARD TOTAL
GDPI	A	13	$38.0	$ 494	-
	B	4	18.0	72	-
	C	12	18.0	216	-
	D	3	5.5	17	-
	E	12	15.5	186	-
	COMSAT	8	6.5	52	-
	TOTAL	52	0	$1037	$58
GNSC	A	9	27.0	243	-
	B	13	9.5	124	-
	C	18	9.5	171	-
	D	5	1.3	7	-
	E	8	27.0	216	-
	TOTAL	53	-	$716	$32
EPS	A	16	10.6	169	-
	B	14	14.4	202	-
	C	6	17.4	104	-
	D	4	6.7	27	-
	TOTAL	40	-	$502	$24
ACS	A	17	1.0	17	-
	B	31	2.0	62	-
	TOTAL	48	-	$ 79	$ 3
	GROSS TOTALS →			$2379	$117

Fig. 2-52 Economic Impact of Application of Standard Subsystems

Mission Coverage	Subsystem	Sub-System Option	RDT&E ($ Million)		Unit ($ Million)		Standard Spacecraft Savings ($ Million)
			Program-Peculiar	Standard Spacecraft	Program-Peculiar	Standard Spacecraft	
15 Missions 151 Spacecraft	CDPI	C	211.5	18.0	307.4	422.8	
	GNSC	C	76	9.5	280.9	280.8	
	EPS	C	212.8	17.4	282.7	350.3	
	ACS	B	29	2	39.1	39.3	
	Struct.	-	61	5.3	65.1	90.6	
	ECS	-	22.0	1.8	25.1	30.2	
	Totals		612.3	54.0	1000.3	1214.0	344.6
23 Missions 219 Spacecraft	CDPI	C	327.50	18.0	479.0	613.2	
	GNSC	E	157	27	598.5	1022.7	
	EPS	C	309.8	17.4	416.1	508.1	
	ACS	B	39	2	56.4	56.9	
	Struct.	-	102.4	7.3	113.1	175.2	
	ECS	-	35.8	3.0	43.0	65.7	
	Totals		971.5	74.7	1706.1	2441.9	161.0

Fig. 2-53 Economic Impact of Application of Typical Standard Spacecraft

2.8.1 System Engineering Approaches

Figure 2-54 lists a few of the low-cost principles which the system engineer might employ.

2.8.2 General Payload Design Guidelines

Figure 2-55 is a listing of the general design guidelines recommended for application to future low-cost payloads.

2.8.3 Low-Cost Subsystem Design

A large percentage of the handbook will be devoted to examples of subsystem design which illustrate the cost-effective design approach. Many of these design examples are listed in Section 5 of this report.

2.9 SUMMARY CONCLUSIONS AND RECOMMENDATIONS

Section 10 of the report provides the conclusions derived from the study and recommendations. A resume is provided following.

2.9.1 Payload Effects in Terms of Cost Reduction

It has been determined by actual preliminary design, preparation of program plans, and costing that payload program savings in the vicinity of 25 percent to 30 percent of the baseline can be obtained by implementation of low-cost techniques. Figure 2-56 shows the unit percentage savings with the low-cost OAO and SEO. With the use of the general-purpose computer (described previously) the OAO savings are increased to about 50 percent. These figures can be extended to combine the costing of payload and launch vehicle and extended over a mission time as shown on Fig. 2-57 for the OAO and Fig. 2-58 for the SEO. The OAO 6-year program, utilizing the Shuttle launch and refurbishment/reuse, indicates a saving of 50 percent of the equivalent costs of a low-cost

- MINIMUM-MANDATORY DESIGN AND PERFORMANCE SPECIFICATION
- SPECIFY ONLY MANDATORY MINIMUM REQUIREMENTS FOR PURCHASED EQUIPMENT
- SPECIFIC MINIMUM DEVELOPMENT AND QUALIFICATION TESTING CONSISTENT WITH HARDWARE RELIABILITY REQUIREMENTS
- TRADEOFF PAYLOAD RELIABILITY VS ON-ORBIT REPAIR/REFURBISHMENT CYCLES TO OBTAIN COST-OPTIMIZED HARDWARE REQUIREMENTS
- TRADEOFF EXPERIMENT VS SPACECRAFT REQUIREMENTS AND ESTABLISH INTERFACE WHICH WILL ALLOW PAYLOAD COST OPTIMIZATION
- TRADEOFF PAYLOAD VS GROUND SUPPORT EQUIPMENT AND SPACE OPERATIONS TO ESTABLISH PROGRAM COST-OPTIMIZED REQUIREMENTS

Fig. 2-54 Low-Cost Payload System Design Approaches

- Design to satisfy mission life and functional requirements - do not overdesign unless there is a cost benefit
- Use qualified off-shelf components where possible
- Apply parts/components for low-stress level operation to obtain added assurance and reduce wearout
- Provide minimum-density installation of equipment to allow easy assembly/removal
- Modularize equipment to allow bench assembly and testing, minimum installation and spacecraft testing, and module replacement
- Provide minimum-density packaging of parts within components and extra volume for growth or modification
- Design for minimum functional complexity
- Design to obtain maximum cost benefit from use of on-orbit checkout and repair/refurbishment

Fig. 2-55 General Low-Cost Payload Design Guidelines

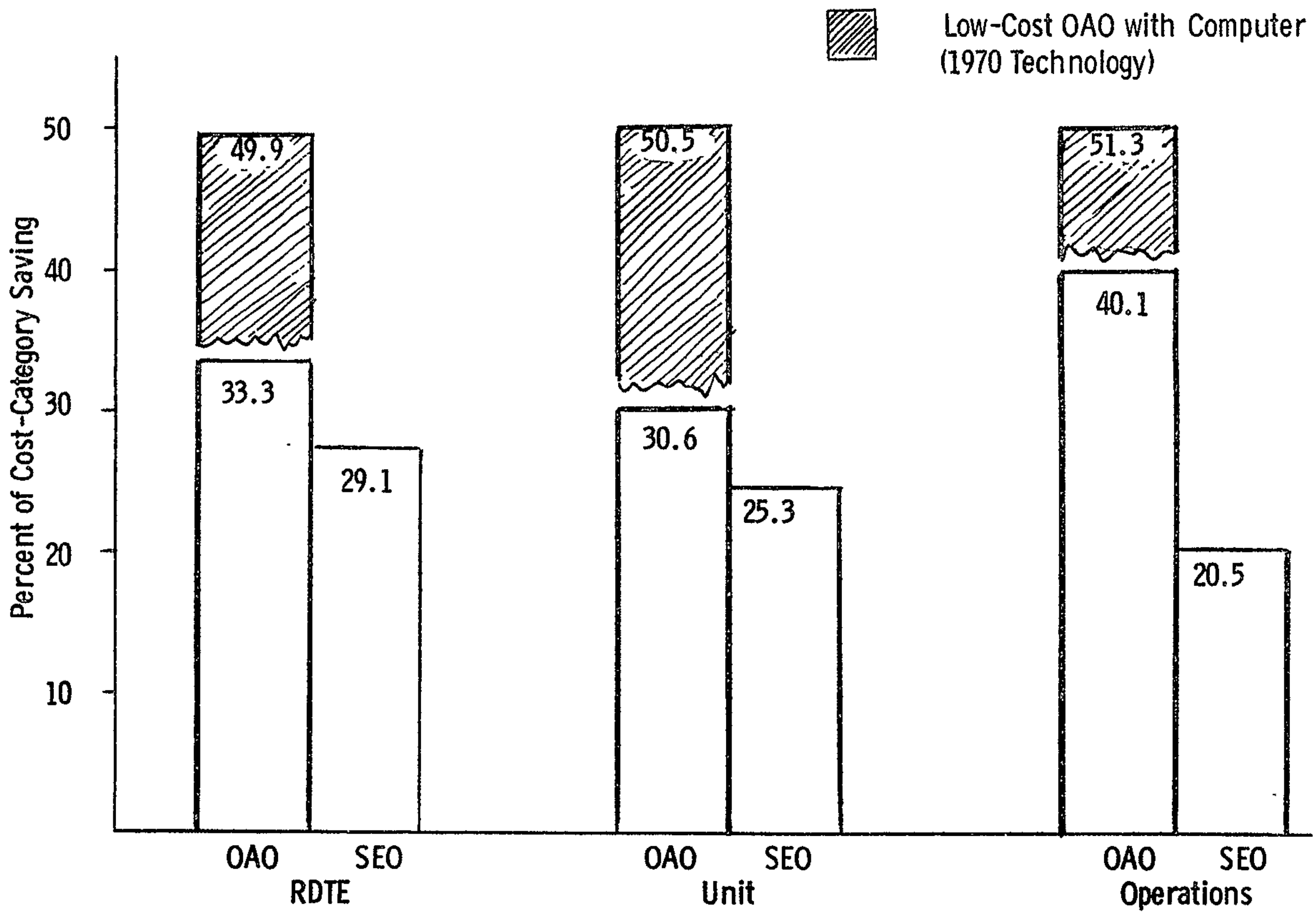


Fig. 2-56 Comparison of OAO with SEO Savings

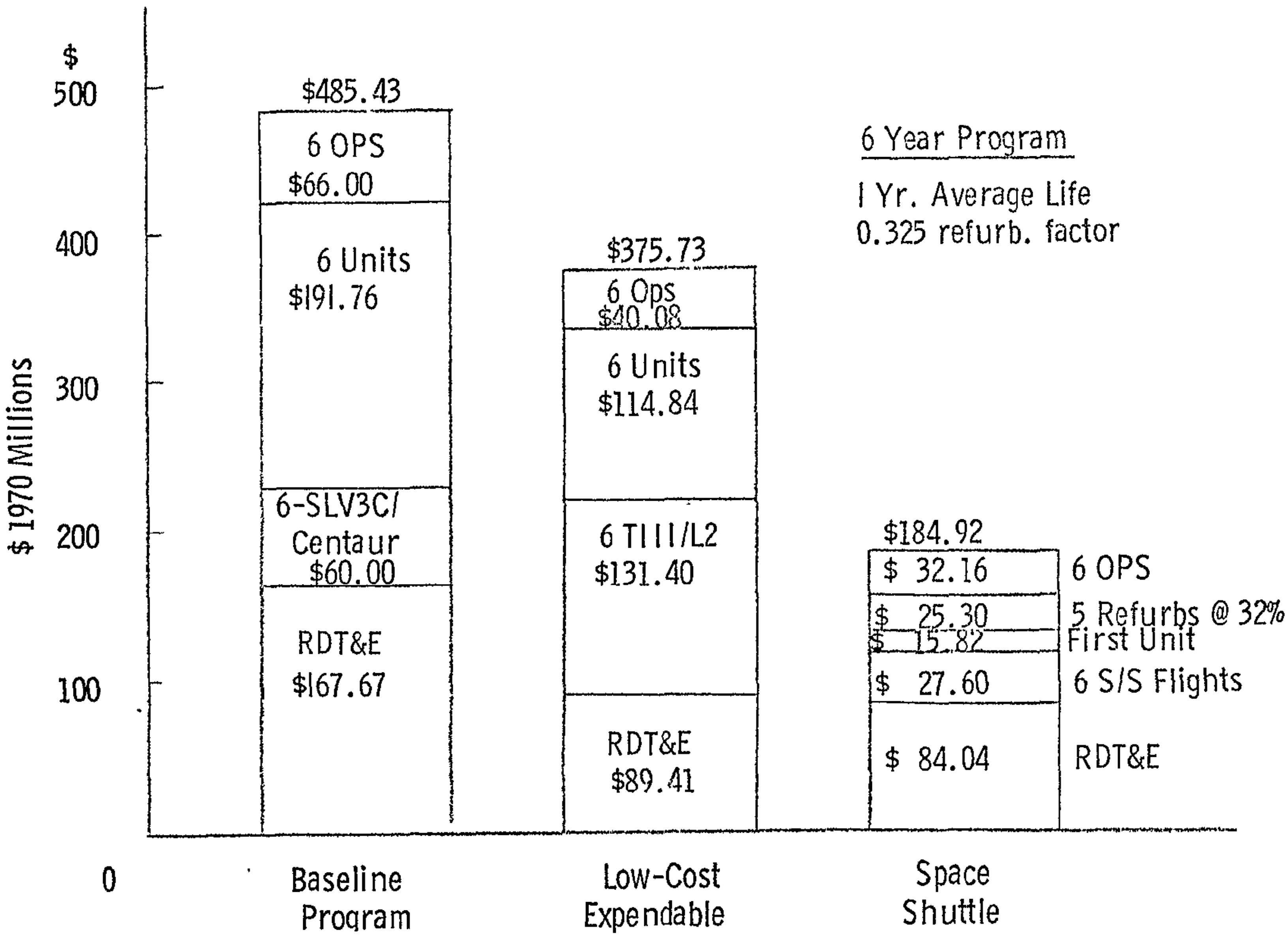


Fig. 2-57 OAO Total Program Cost Comparison

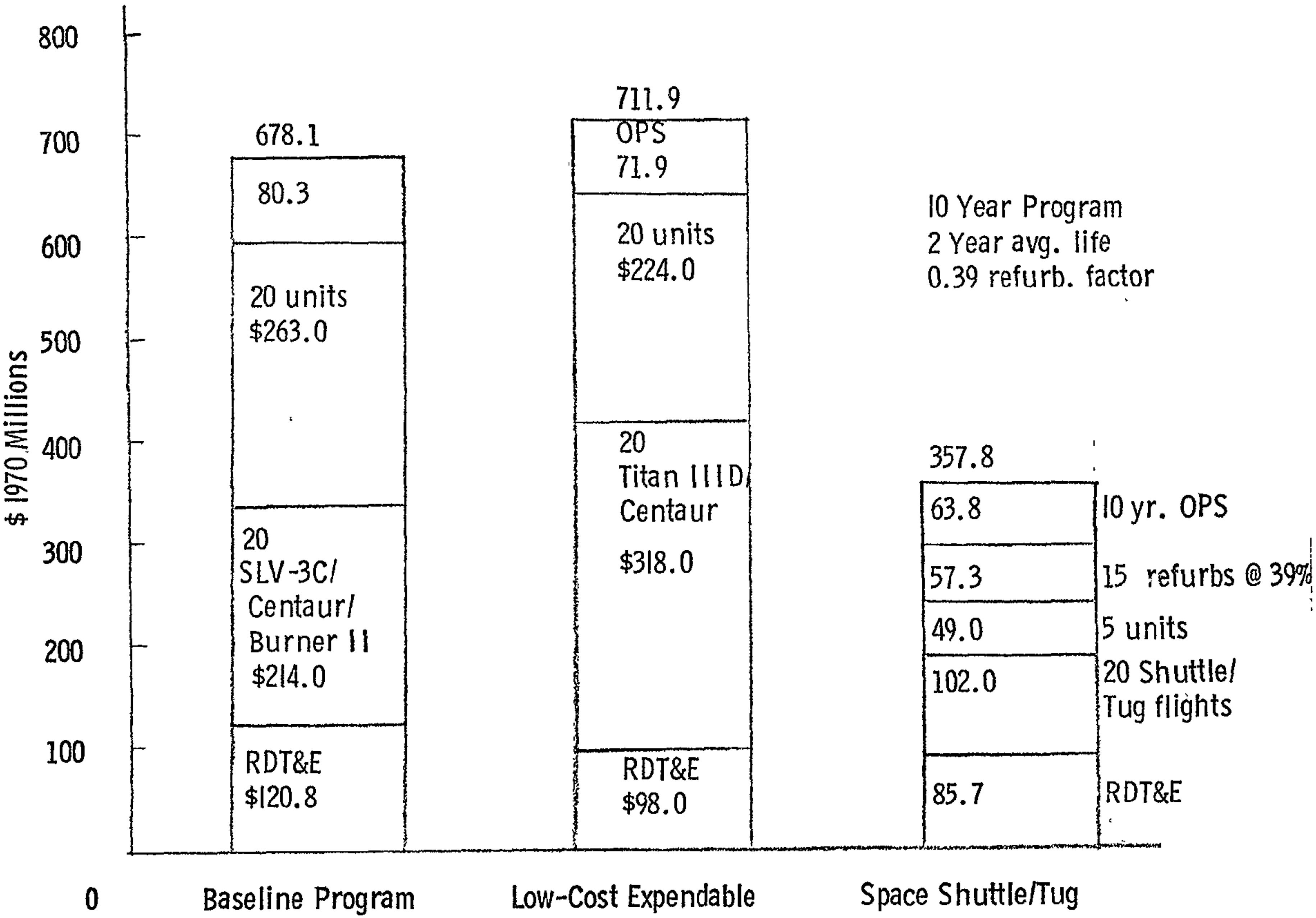


Fig. 2-58 SEO Total Program Cost Comparison

payload launched on a low-cost expendable. Similarly, on a 10-year SEO program, a 50 percent saving is possible using a Shuttle-launched low-cost payload in lie of a low-cost expendable-launched.

2.9.2 Effect of NASA Policy on Cost-Reduction

At the request of NASA/HQ, a brief survey was made of NASA operational policies as they might affect implementation of low-cost payloads. The items listed on Fig. 2-59 were provided. The cost of these policy changes has not been quantified; rather, they were intended only as a broad-spectrum "beginning" list. It is assumed that NASA will implement whatever follow-on analysis they determine to be appropriate.

2.9.3 Basic Conclusions from the Study

The principal conclusions derived from the Payload Effects Study are listed on Fig. 2-60.

2.9.4 Additional Cost Reduction Areas

Figures 2-61a and 2-61b provide a list of recommendations relevant to extending the payload cost-reduction concepts to obtain even further program savings.

POLICY CHANGE	REASON AND/OR EFFECT
• Reduce degree of configuration management (hardware traceability)	• Failures can be determined by actual inspection of retrieved hardware
• Utilize standard subsystems and checkout equipment (in lieu of project peculiar)	• Large savings in RDT&E costs. Allows reduction also in training and operations costs.
• Reduce amount of contract documentation	• Changes possible in NASA procurement documents.
• Reduce reliability and QA requirements and reduce acceptance tests • Use lower-grade (perhaps aircraft quality) hardware	• Ability for in-orbit checkout and repair allows higher risk on payload hardware
• Shorten program time (spacecraft and experiments)	• Flight-testing hardware on Shuttle sortie missions (flying lab) allows shortening of devel/qual. test.
• Increased autonomy of payload/Shuttle flight operations	• Allows reduction of ground support facilities and personnel.
• Standardize missions and consolidate experiment objectives (orbit inclination, altitudes, etc.)	• Allows reduction in variety of spacecraft and multi-payload launch and revisit for Shuttle.

Fig. 2-59 NASA Policy Issues Affecting Program Costs (for Shuttle-Launched Payloads)

- The Payload Effects Study has confirmed that significant cost benefits accrue from new payloads designed to low-cost criteria:
  - Reduced weight/volume constraints
  - Modularization and repair/refurbishment/reuse
  - Relaxed reliability requirements with higher risk (Shuttle only)
- Additional significant savings, primarily in RDT&E, are possible with standardization of payload subsystems and/or with use of standard spacecraft
- The savings developed in this study for low-cost payloads are conservative; as Shuttle flight experience is gained, additional cost savings are forecast.
- Some of the payload savings can be implemented prior to the Shuttle era on current expendables
- With planned NASA budget limits, it is important to continue vigorous pursuit of payload cost reduction in order to provide the $ savings which will make the Shuttle itself a reality.

Fig.2-60 Conclusions of Payload Effects Study

LMSC-A990556

RECOMMENDED APPROACH	COST REDUCTION EFFECTS
• Standardize unmanned payload subsystems	• Large savings in RDT&E
• Standardize experiment interfaces	• Allows standard refurbishment depots and lower refurbishment costs
• Utilize minimum quantity of multi-mission standard spacecraft	• Allows reduced-cost training of field crews in standard approaches to payload hardware repair, refurbishment, checkout
• Standardize unmanned payload checkout equipment - for ground and in-orbit usage	• Allows simplification of Shuttle interfaces with payloads and standardization of cargo crew operations procedures
	• Allows simplification of ground support facilities and uniformity of support personnel across many projects
• Apply low-cost design approaches to other payloads: Shuttle, Space Tug, manned payloads, lunar mission hardware, etc.	• Dollars saved on unmanned payloads can be applied to enlarging the total space exploration capability (e.g., inclusion of lunar programs). Further similar savings can be obtained from other space payloads.

Fig. 2-61a Additional Technical Considerations for Cost Reduction (Sheet 1 of 2)

RECOMMENDED APPROACH	COST REDUCTION EFFECTS
• Apply new technology only after thorough cost-effectiveness tradeoff assumes lower program costs	• Reduced risk (and cost) in new programs schedules and assurance that cost-optimized hardware is adopted
• Implement analyses to cost-optimize the combined effects of payload subsystem reliability and operating life vs launch ascent effects vs orbit repair and refurbishment	• Provides minimum-cost payload • Provides guide for detail provisioning of spare modules, components, and parts for each payload (procured with payload) • Establishes base for payload field repair/refurbishment depot implementation
• Conduct more detailed analyses of repair and refurbishment of typical unmanned payloads and extend to other types of payloads	• Refurbishment of payloads is the single largest cost-driver and any further reduction in refurb-to-new cost ratio can be multiplied by large quantities of payloads in the overall Mission Model.

Fig. 2-61b Additional Technical Considerations for Cost Reduction
(Sheet 2 of 2)

Section 3
BASELINE PAYLOAD SELECTION AND DESCRIPTION

The selection of baseline payloads that are representative of typical existing NASA unmanned satellites represents the essential first step of the Payload Effects Analysis Study logic. To provide credibility to the study results, it was mandatory that detailed design and cost data were available for the chosen payloads. This section presents the rationale for selection, a historical survey of events leading to the eventual selection of the baseline payloads, and summary descriptions of each including cost, weight and reliability data which was used in the parametric analyses.

3.1 PAYLOAD SELECTION

3.1.1 Rationale

The NASA guidelines for selection of the baseline payloads were as follows:

- Reliable requirement and cost data shall be available for current examples of similar satellites to provide the basis for cost comparison.

- The satellites selected shall cover the range of costs, based on current experiences; e.g., sophisticated satellites, made necessary by demanding mission requirements, resulting in high cost per pound at one end of the range, and unsophisticated low cost satellites at the other. Specifically, the following requirements should be satisfied in the selection:

  (1) Sophisticated satellites generally associated with high program dollar value per pound in orbit (i.e., physics/astronomy).

(2) Medium cost satellites generally associated with first generation or development satellites for potential space applications.

(3) Operational satellites such as weather and communications considered low cost program devices.

- The selected payloads shall have actually flown or shall be similar to ones that have flown."

3.1.2 Selection Process

As a result of a comprehensive review of past and future scientific space programs, LMSC in its Technical Proposal for the Payload Effects Analysis Study, suggested the Lunar Orbiter, the Gemini/Agena Target Vehicle (GATV), and the Small Research Satellite (SRS, the LMSC P-11 Subsatellite) as baselines. In coordination with NASA the Orbiting Solar Observatory (OSO) was substituted for GATV as being more representative of current and future satellite payloads. However, after initiation of the study it was found that historical data on OSO were not readily available. Thus, following agreement with Goddard Space Flight Center, the Orbiting Astronomical Observatory (OAO-B) was substituted for OSO. As a result of this change, the Lunar Orbiter became the intermediate class spacecraft, and the OAO became the representative of the more expensive/ complex type.

Inspection of the mission model, provided by NASA and Aerospace, showed no lunar orbit unmanned payload missions projected for the period of application of the combined study results (1978-1990), and a question arose as to the direct suitability of extending Lunar Orbiter to future low-cost programs. However, comprehensive cost and design data on the Lunar Orbiter were available and there were no alternatives that fulfilled the requirements of data availability and security classification. It was, therefore, agreed that LMSC examine the use of Lunar Orbiter derivatives for the performance of more representative missions. It was readily apparent that unmanned synchronous equatorial and planetary missions, which represent a significant portion of the NASA mission model,

should be reflected in the baseline payloads. Furthermore, the Lunar Orbiter and its experiment payload can be viewed as being representative in cost and degree of complexity to spacecraft specifically developed for these missions.

Hence it was decided that the Lunar Orbiter design be extrapolated for application as a 1-year Synchronous Equatorial Earth Resources Observation Satellite (SEO) and a Mars Orbiter (MO) retaining as many as possible of the original Lunar Orbiter elements. Primary emphasis was directed towards the SEO configuration with backup data to be provided at a secondary level of emphasis on the MO. During subsequent studies, NASA requested that the SEOs life be extended to 2 years for better correlation with the NASA Mission Model.

Selection of the SRS was approved as being representative of the family of small space physics satellites such as Explorer and Pioneer. Thus, the selection process was completed and approved by NASA Headquarters as follows:

Payload Class	Payload Selection
Low Cost/Complexity	Small Research Satellite (SRS)
Medium Cost/Complexity - Primary -	Synchronous Equatorial Earth Resources Observatory (SEO) (1 and 2 year lifetimes) Lunar Orbiter Derivative
Medium Cost/Complexity - Secondary -	Mars Orbiter (MO) - Lunar Orbiter Derivative
High Cost/Complexity	Orbiting Astronomical Observatory (OAO-B)

Selection of these payloads assured NASA of specific low-cost data on a representative portion of the future mission model. OAO furnishes the opportunity to evaluate the payload effects of revisit, maintenance, and refurbishment of highly complex systems; SEO permits examination of the use of the Shuttle/Tug combinations during earth synchronous orbit missions; MO permits study of the effects of new transportation systems on complex one-way earth escape payloads; and SRS represents low-cost expendable payloads.

3.1.3 Payload Subsystem Definition and Hardware Breakdown

The nomenclature applied to hardware elements of various payloads is similar, yet sufficiently different to cause misinterpretations in detail comparisons. Also, the specific types of components included in a particular subsystem were noted to vary from payload to payload. A standard system of reference was, therefore, developed so that costs could be collected and analyzed on a consistent basis relevant to the hardware.

Subsystem categories were established as follows to subdivide the payload into eight elements:

- Experiments
- Structures and Mechanisms
- Environmental Control
- Electrical and Pyrotechnics
- Guidance and Navigation (Stabilization and Control)
- Propulsion and Attitude Control
- Telemetry, Tracking and Command (Communications, Data Processing and Instrumentation)
- Adapter

Figure 3-1 provides a description of each subsystem and lists the typical hardware included in each. This "standard" breakdown, approved by NASA for this study will be utilized to identify all elements of both the baseline and the low-cost payload subsystems. All costs and design characteristics will be correlated with these subsystems.

Summaries of the baseline payload subsystem characteristics are provided in Figs. 3-2 and 3-3.

SUBSYSTEM ELEMENT	TYPICAL HARDWARE	SUBSYSTEM ELEMENT	TYPICAL HARDWARE
PAYLOAD ASSEMBLY & INTEGRATION (All elements which are part of the payload system but external to payload assembly)	• SEPARATION DEVICES • PAYLOAD ADAPTERS & INTERSTAGES • FAIRINGS (NOT STD.EXIT) • UMBILICALS • SAFETY DEVICES	ELECTRICAL AND PYROTECHNICS (All elements of electrical power generation, control, distribution. Also pyrotechnic hardware.	• BATTERIES • SOLAR ARRAYS (INCL STRUCTURAL PANELS, SOLAR CELLS, DIODES, INTERCONNECTS, ORIENTATION ASSY. • VOLTAGE REG., INVERTERS, • DISTRIB., PRIMARY & INST. CABLING • PYROTECHNIC DEVICES (SQUIBS, ETC.)
EXPERIMENTS (All elements which are mission-peculiar and not part of the supporting spacecraft. Includes any data processing equipment which is integral with experiments.)	• TELESCOPES • CAMERAS • TV CAMERAS • PHYSICS EXPERIMENTS • RADIOMETERS SPECTROMETERS, etc.	GUIDANCE, NAVIGATION, STABILIZATION, CONTROL (All elements which provide flight control, orbit positioning, and attitude hold, but excluding thruster system.)	• POSITION SENSORS (SOLAR, EARTH, STAR) • MOMENTUM WHEELS • FLIGHT CONTROL ELECTRONICS • GYROS • INERTIAL REF. UNITS
STRUCTURES, MECHANISMS & VEHICLE ASSEMBLY (All structural & mechanical elements which are not part of the other 6 functional subsystems. Also includes install. of subsystems into spacecraft & attachment of experiment.)	• SPACECRAFT STRUCTURE • EQUIPMENT SUPPORTS • SUN BAFFLES • BALANCE BOOMS & EXTNS.MECH. • ANTENNA DEPLOY. MECH. • SOLAR ARRAY DEPLOY. MECH.	PROPULSION (All elements which are provided for major changes in velocity vectors ATTITUDE CONTROL Elements for control and/or maintenance of attitude which involve mass expulsion.	• SOLID-PROPELLANT MOTORS COLD GAS, MONOPROPELLANT, OR BI-PROP. THRUSTERS • TANKAGE FOR PROPELLANT, COLD GAS, PRESSURANTS • PLUMBING AND VALVES • PROPELLANT
ENVIRONMENTAL CONTROL (All elements which alter and/or control the temperature of the payload and components thereof.)	• THERMAL LOUVERS • INSULATION • THERMAL PAINTS & COATINGS • THERMOSTATS • HEATERS • RADIATORS, HEAT PIPES	TELEMETRY, TRACKING, & COMMAND (All elements of Data Processing, Instrumentation, Telemetry, Communications & Command)	• DATA HDLG., PROCESSING, STORAGE EQUIPMENT • SIGNAL CONDITIONERS • TRANSDUCERS • XMITERS, BEACONS, XPONDERS • RCVRS/DECODERS • MULTIPLEXERS/ENCODERS • ANTENNAS • RF POWER AMPLIFIERS • CMD.DATA STORAGE, TIMING

Fig. 3-1 PAYLOAD SYSTEM HARDWARE BREAKDOWN

SUBSYSTEM	ORBITING ASTRONOMICAL OBSERVATORY	SMALL RESEARCH SATELLITE
EXPERIMENTS AND MISSION-PECULIAR EQUIPMENT	38 APERTURE CASSEGRAIN TELESCOPE WITH ASSOCIATED ELECTRONIC PACKAGES FOR STELLAR UV OBSERVATION IN 1100 Å TO 4000 Å REGIME.	SPACE PHYSICS EXPERIMENT PACKAGE FOR IONOSPHERIC MEASUREMENTS OF PARTICLES AND ENERGIES.
STRUCTURES AND MECHANISMS AND THERMAL CONT.	OCTOGONAL BOX STRUCTURE WITH CENTRAL TUBULAR COMPARTMENT TO ACCEPT EXPERIMENT. LARGE FIXED SUN SHIELDS ON ONE END FOR TELESCOPE AND BORESIGHT STAR TRACKER LIGHT SHIELDING DEPLOYABLE SOLAR ARRAY PANELS. DEPLOYABLE BALANCE BOOMS. PASSIVE T.C. WITH THERMOSTAT-CONTROLLED LOUVERS.	3-BAY SIMPLE BOX STRUCTURE. 4 DEPLOYABLE SOLAR ARRAY PANELS. PASSIVE THERMAL CONTROL.
GUIDANCE AND NAVIGATION	COARSE POINTING WITH COARSE WHEELS TO PLUS OR MINUS 30 ARC SEC WITH DRIFT RATE OF LESS THAN 15 ARC SEC IN 50 MIN. FINE POINTING WITH FINE WHEELS WITHIN 0.1 ARC SEC USING EXPERIMENT FINE ERROR SENSOR. MOMENTUM DUMPING WITH ELECTRONIC TORQUERS OR N_2 GAS JETS. SENSORS: STAR TRACKERS FOR COARSE POINTING, SUN SENSORS FOR ACQUIS. AND SUN-BATHING MODES, GYROS FOR RATE SETTLING, ATTITUDE HOLD, SLEWING.	WOBBLE DAMPER. FIXED-AXIS OPERATION AFTER INITIAL SPIN-UP.
ATTITUDE CONTROL	N_2 COLD GAS: PRIMARY AND SECONDARY THRUSTER SETS.	60 TO 85 RPM SPINUP WITH 2 SOLID ROCKETS
TT&C	2 WIDEBAND T/M XMTRS @ 400 MHz; 2 NARROW BAND XMTRS @ 136 MHz. 2 TRACKING BEACONS @ 136 MHz. 4 COMMAND RCVRS @ 148 MHz. CORE MEMORY WITH TAPE RECORDER FOR CONT. S/C DATA. COMPATIBLE WITH NASA STADAN.	VHF FM/FM T/M. TONE/DIGITAL COMMAND. TAPE RECORDING.
PROPULSION	NO PRIMARY PROPULSION.	DUAL SOLID MOTORS FOR ORBIT ATT.
ELECTRICAL	SOLAR ARRAYS AND NiCd BATTERIES-60 AH SUPPORTS EXPERIMENTS: 60W PEAK, 30W AVG.	20W CONTINUOUS FROM SOLAR ARRAYS AND BATTERIES AND 50W AT 15% DUTY CYCLE.

Fig. 3-2 SUMMARY OF BASELINE PAYLOAD SUBSYSTEM CHARACTERISTICS - OAO AND SRS

SUBSYSTEM	LUNAR ORBITER (REFERENCE)	SYNC. EQ. EARTH RESOURCES	MARS ORBITER
EXPERIMENTS & MISSION PECULIAR EQUIPMENT	HIGH/MEDIUM RESOLUTION PHOTOGRAPHY. ON-BOARD FILM PROCESSING AND WIDEBAND DATA READOUT CAMERA RESOL. AT 46 KM = 1 METER. V/H SENSOR FOR IMC.	SAME AS LUNAR ORBITER EXCEPT: - ADD ADV.-VIDICON CAMERA SYSTEM (TV) - DELETE V/H SENS. & MED. RES. LENS - INCREASE FILM QUANTITY - CHANGE READOUT METHOD & ADD 2nd OMS	SAME AS LUNAR ORBITER EXCEPT: REDUCED RESOLUTION ~ 30M FROM 110 NM ADD U/V & IR EXPERIMENTS FROM MARINER '71 AND 2nd OMS
STRUCTURES & MECHANISMS & THERMAL CONTROL	OPEN PLATE AND TRUSS STRUCTURE. PASSIVE THERMAL PROTECTION; CONTROLS TEMPERATURE WITHIN 35° TO 85°F. EXTENDABLE SOLAR ARRAYS AND ANTENNAE.	SAME AS LUNAR ORBITER EXCEPT: - STRENGTHEN TANK SUPPORT STRUCTURE - ADD MOUNTING FOR TV CAMERA - ADD FILM SUPPLY SHIELDING	SAME AS LUNAR ORBITER EXCEPT: - STRENGTHEN TANK SUPPORT STRUCT. - ADD HEATERS & INSULATION - ADD PROVISIONS FOR SECONDARY EXPERIMENTS - INCREASE CAPABILITY OF DEPLOY. MECHANISMS
GUIDANCE & NAVIGATION	5 SUN SENSORS FOR P/Y REF.; CANOPUS TRACKER FOR ROLL. INERTIAL REF. UNIT USED WHEN SUN/STAR NOT VISIBLE. CAMERA POINT WITHIN PLUS OR MINUS 0.2 DEG. V/H SENSOR.	SAME AS LUNAR ORBITER EXCEPT: - ADD EARTH HORIZON SENSOR - ADD 2 REACTION WHEEL & CONTROLS - DELETE CANOPUS TRACKER - ADD PITCH WHEEL & PITCH RATE GYRO	SAME AS LUNAR ORBITER EXCEPT: - REPLACE IRU WITH IMPROVED UNIT WITH HIGHER RELIABILITY AND LONGER LIFE.
ATTITUDE CONTROL	THREE-AXIS ACTIVE. 8 GN_2 THRUSTERS	SAME AS LUNAR ORBITER EXCEPT: INCR. N_2 SUPPLY AND ADDED REDUNDANT PLUMBING & VALVES ADDED E-W THRUSTERS	SAME AS LUNAR ORBITER EXCEPT: - ADD ADDITIONAL N_2 TANK + PLUMBING - DOUBLE N_2
TT & C	HIGH & LOW GAIN ANTENNAS. DIGITAL COMMAND & PROGRAMMER, TRANSPONDER, TRACKING & DATA STORAGE. TRANSMISSION TO EARTH AT 40 MIN./EXPOSURE. 10W & 0.5W TRANSMITTERS	SAME AS LUNAR ORBITER EXCEPT: - ADD TAPE RECORDER & ELECTRONICS - CHANGE TO FIXED HIGH GAIN ANT. - ADD REDUNDANT TWTA & PCM MULTIPLEXER/ENCODER	SAME AS LUNAR ORBITER EXCEPT: - ADD TAPE RECORDER + ELECTRONICS - REDUNDANT TRANSPONDER + PCM MUX/ENCODER - REPLACE 10W TWT WITH 2 40W - REPLACE PARABOLIC HI-GAIN ANT. WITH 9' DIA. ANTENNA
PROPULSION	BIPROPELLANT PRESSURE-FED; 100 LB THRUSTER FOR MIDCOURSE AND LUNAR RETRO MANEUVERS	SAME AS LUNAR ORBITER EXCEPT: - REPLACE 4 BIPROP TANKS WITH ONE COMMON-BULKHEAD LARGER TANK - ADD BIPROP FOR ORBIT INSERTION	SAME AS LUNAR ORBITER EXCEPT: - REPLACE 4 BIPROP TANKS WITH ONE COMMON-BULKHEAD LARGER TANK - ADD BIPROPELLANT FOR TRANS-MARS MIDCOURSE AND FOR MARS RETRO
ELECTRICAL POWER	SOLAR ARRAYS (10856 CELLS) + NiCd SECONDARY BATTERY - SUPPLIES 375 W.	SAME AS LUNAR ORBITER EXCEPT: - ADD ONE CHARGE CONT. + SHUNT REG. - ADD ONE BATTERY - ADD SUN TRACKER - ADD SOLAR ARRAY ORIENTATION DEVICE (4)	SAME AS LUNAR ORBITER EXCEPT: - INCREASE SOLAR ARRAY AREA - ADD BATTERY - ADD REDUNDANT SHUNT REG. + CHARGE CONTROLLER

Fig. 3-3 LUNAR ORBITER DERIVATIVES

3.1.4 Baseline Payload Cost Apportionment

The cost analysis support of the Cost Optimization Analysis, consisted of collection, review, and synthesis of the baseline payload historical cost data. Historical data were collected for the Small Research Satellite (SRS) or P-11, the Orbiting Astronomical Observatory (OAO-B), and the Lunar Orbiter (LO) programs. Although the Lunar Orbiter was not used per se, it constituted a reference payload from which two derivatives were extrapolated, the Synchronous Equatorial Earth Resources Satellite (SEO) and the Mars Orbiter (MO). The latter two payloads formed simulated baselines with simulated "historical" costs and design state-of-the-art representative of the 1964-67 time period as extrapolated from the actual Lunar Orbiter reference program.

3.1.4.1 Cost Allocation Approach. The objective of the cost allocation effort was to assign historical cost data to the baseline subsystems and cost categories standardized for this study, and to extrapolate simulated historical costs for the SEO and MO programs. The cost was broken down into the basic categories of non-recurring or RDT&E cost and the recurring unit hardware cost, and operations cost. Allowing for the quantities of flight articles and mission operations period for each baseline program, total program cost was summed from above categories. For backup information, RDT&E costs were further broken down into the functional cost categories, such as development, GSE, spacecraft integration and test, and program management, by subsystem. Unit hardware cost and operations costs were broken down only by subsystem. Costs which could not be allocated by subsystem (e.g., transportation) were aggregated into a non-allocatable cost category.

The approach to accomplishing this task was to obtain a complete (as possible) set of historical cost data and to check their breakdowns and contents to assure consistency with the desired Payload Effects Study cost category breakdowns. Any incompletions and inconsistencies were adjusted and the costs were then allocated by subsystem to provide a standardized set of baseline payload data.

3.1.4.2 Unmanned Payload Cost Model. The unmanned payload cost model represents the baseline payload cost aggregation. The elements of this model, and the manner in which they are summed for the total program, are as follows:

$$TPC_P = \left[(NRC_P + RC_P)\ CF \right] PM$$

TPC_P = Total Payload Program Cost

NRC_P = Total Non-Recurring Payload Cost

RC_P = Total Recurring Payload Cost

CF = Prime Contractor Fee*

PM = Customer Program Management*

$$NRC_P = DC_P + GC_P + IC_P + MC_P$$

DC_P = Development Cost of Payload

$$DC_P = \left[\sum_{i=1}^{j} DC_i \right] + UD$$

DC_i = ith Subsystem Development Cost

UD = Unallocated Development Support

j = No. of Subsystems in the Payload

GC_P = Payload GSE Cost

IC_P = Spacecraft Integration and Test Cost

MC_P = Program Management Cost

$$RC_P = Q \left[\sum_{i=1}^{j} HC_i + SOC + EOC \right]$$

Q = Quantity of Flight Units in the Program

HC_i = ith Subsystem Unit Hardware Cost

SOC = Spacecraft Unit Operations Cost

EOC = Experiment Unit Operations Cost

* This factor was not included in the analyses of this study.

Definitions - To clarify the content of the various cost categories, the following definitions are provided:

Payload - Describes the spacecraft and experiments but excludes launch vehicle elements.

Subsystems - These are as previously defined.

Subsystem Development Cost - Includes development, design, engineering, tooling, developmental materials, and support for a given subsystem.

Unallocated Development Support - Includes costs not allocated to a subsystem and includes mission planning, launch and flight operations planning, logistics, facilities support, QA and reliability support, and computer programs related to total mission.

GSE Cost - Consists of all GSE, AGE, and test equipment including GSE for the experiments and mission peculiars. (Since payload quantities per given program have been historically small, all payload GSE development and procurement is charged to the non-recurring cost).

Payload System - Describes the payload, payload-launch vehicle adapter and any payload peculiar additions to the launch system.

Spacecraft Integration and Test Cost - Includes component and subsystem test, test program integration and sustaining engineering, system analysis and integration, and test data reduction.

Program Management - Includes administrative and support costs associated with project management office. (Recurring hardware and operations costs include recurring program management and sustaining engineering costs.)

Hardware Unit Cost - Consists of the sum of the subsystem hardware costs including spares and replacement parts prorated to the flight unit. (Since quantities of the same payload hardware have been historically small, no learning curve has been identified or applied in this model.)

Operations Costs - Include launch and flight operations, logistics support, flight data collection, reduction and reporting, sustaining engineering and integration, GSE support and maintenance, relevant to the payload.

3.1.4.3 Cost Apportionment Assumptions. In organizing and standardizing the baseline payload cost data, several assumptions were made to maintain consistency within the data sets. These assumptions are:

- Payload system costs do not include shrouds, unless specifically developed for the payload.
- Payload program costs do not include launch vehicles.
- The Adapter Subsystem includes primarily the costs of adapter and its integration with the launch vehicle. The structural integration of the spacecraft and experiments is charged to the Structure and Mechanisms Subsystem.
- Prime Contractor's fee (CF) has not been included, although the subcontractors fees on experiments and other subsystems are included.
- Customer's program management costs (PM), such as program office, etc., are not included in the baseline payload program costs.
- Learning curve effects, if any, were not considered in the baseline.
- Since the payloads were built over a time span from 1961 to the present, the historical cost data were adjusted for inflation and converted to 1970 dollars. In the conversion, the following Inflation Factors, based on the Aerospace Guided Missile and Spacecraft Inflation Index, were used:

Payload Item	Cost	Year Span Cost Data Base	Inflation Factor for 1970 $ Base
OAO Spacecraft	RDT&E	1961-1968	1.3721
	Fab.	1969-1970	1.0315
	Ops.	1970	1.0000
OAO Experiments	RDT&E	1967-1969	1.1390
	Fab.	1969-1970	1.0315
	Ops.	1970	1.0000
Lunar Orbiter (SEO & MO Derivatives)	RDT&E	1964-1966	1.33073
	Fab.	1965-1967	1.26537
	Ops.	1966-1967	1.23350
Small Research Satellite	RDT&E	1961-1965	1.48438
	Fab.	1968	1.144
	Ops.	1968	1.144

All baseline payload cost data given throughout the body of this report are in 1970 dollars.

3.1.5 Baseline Payload Weight and Reliability Apportionment

To provide a consistent base for computer-comparison and tradeoff of the weights, reliabilities, and costs of the four selected payloads, a reference set of data were established on each payload to the subsystem level. An analysis was made of reference data available on each payload: weight statements, reliability reports, design specifications, and other data pertinent to hardware functional description. For clarification of hardware functional redundancy arrangements, direct contacts were made with the payload project offices (Boeing and Kodak on the Lunar Orbiter; NASA/Goddard on the OAO; LMSC P-11 Program on the SRS). All reliability characteristics used during the study were based upon estimates made of the probability that the payload and parts thereof will perform satisfactorily without catastrophic failure. Actual performance figures on operational payloads, even though available in some cases, have not been used. In this way, the uncertainties created by "partial-success" missions has been avoided and a common reference base has been maintained for cost-optimization.

3.1.6 Launch Vehicle Costs, Reliability

3.1.6.1 Launch System Selection. With NASA and Aerospace Corporation agree· ment, a launch vehicle system was selected for each of the payload types in three categories: alternate current expendable, low-cost expendable, and re· usable. Figure 3-4 identifies the launch vehicle systems.

3.1.6.2 Costs of Launch Systems. Costs for these launch systems were provi(by the Aerospace Corporation; these costs, representing the best estimates o: costs extrapolated to the 1978-1990 time period, were based upon (1) actual (rent launch systems, (2) estimates of the low-cost expendable future systems, and (3) on estimates of the Space Shuttle operational costs. By agreement a: NASA, Aerospace, Mathematica, and LMSC, the cost per launch of the Shuttle w: based upon an assumed marginal cost, excluding amortization of the Shuttle ir vestment. These user's costs are tabulated in Fig. 3-5 for each of the selec ted launch vehicle systems. The use of these costs, or apportioned percenta£ thereof, in the cost-optimization analysis is explained later in this report.

3.1.6.3 Launch System Reliability. Estimates of launch system reliability v made by LMSC based upon (1) historical experience with expendable systems, a: (2) estimates of Shuttle and Space Tug reliability coordinated by Aerospace v NASA agencies. Agreement on reliability characteristics was reached by NASA, Aerospace, and LMSC. The numbers are also listed for each launch system in Fig. 3-5.

These numbers represent the probability of successful performance of the lau: system up to the point of separation into free flight of the payload. Becau: the jettison of the payload fairing and the separation of the payload from tl launch vehicle or upper stage are so functionally oriented to the payload, tl reliability of these two operations has been included with the payload syster rather than with the launch system. The launch system reliability numbers w(later entered directly into the cost-optimization runs for the parametric analysis.

PAYLOAD TYPE	MISSION DESTINATION	CURRENT ALTERNATE EXPENDABLE	NEW LOW-COST EXPENDABLE	REUSABLE
OAO	400 NM; 35°	ATLAS/CENTAUR	7 SEG SRM (2)/ TITAN LDC (TIII-L2)	SHUTTLE (57000 LB to 100 NM; 28.5°)
MARS ORBITER	998 NM x 24 NM MARS ORBIT	TITAN IIID/CENTAUR	TITAN IIID/CENTAUR	SHUTTLE (72200* LB to 100 NM; 28.5°)/ SPACE TUG
SYNC. EQ. EARTH RESOURCES	19320 NM; 0°	TITAN IIID/CENTAUR	TITAN IIID/CENTAUR	SHUTTLE (72200** LB to 100 NM; 28.5°)/ SPACE TUG
SRS	300 NM; 82°	ATLAS SLV-3C/BURNER II (10 FT. DIA.)	3 SEG SRM/TITAN CORE II/ AGENA (10 FT. DIA.)	SHUTTLE (41000 LB to 100 NM; 28.5°)

1 nm = 1.852 km
1 lb = 0.4536 kg

* Space Tug delivers payload into trans-Mars trajectory and returns to LEO.

** Space Tug delivers payload to Sync. Eq. orbit and returns empty to LEO or returns empty to LEO or returns with equal-weight payload.

Fig. 3-4 Selected Launch Vehicle/UpperStage/Payload Combinations

LAUNCH VEHICLE/ UPPER STAGE	L.V. CATEGORY	COST PER FLIGHT ($ MILLION)			REL. CHAR.	PAYLOAD ENVELOPE LIMIT (INCHES)	PAYLOAD DELY. CAPABILITY		
		RECUR	OPS	TOT *			TYPE	WT.(LB)	ORBIT
ATLAS SLV-3A/BURNER II	ALT. CUR.	3.5	1.3	4.8	.950	120D	SRS	4000	300 NM; 82°
ATLAS SLV-3C/CENTAUR	ALT. CUR.	6.3	2.3	8.6	.936	120D	OAO	9500	400 NM; 35°
TITAN IIID/CENTAUR	ALT. CUR.	12.0	3.0	15.0	.945	120D	MARS ORB.	7000	MARS INJECT
	ALT. CUR.	12.0	3.0	15.0	.945	120D	ERS	7000	SYNC. EQ.
	LOW COST	12.0	3.0	15.0	.945	180D	ERS	7000	SYNC. EQ.
	LOW COST	12.0	3.0	15.0	.945	180D	MARS ORB.	7000	MARS INJECT
3 SEG SRM/TITAN CORE II AGENA	LOW COST	5.0	1.5	6.5	.968	120D	SRS	4000	300 NM; 82°
7 SEG SRM (2)/ TITAN LDC	LOW COST	15.0	3.0	18.0	.985	180D	OAO	10000	400 NM; 35°
SHUTTLE (57000 LB TO 100 NM; 28.5°)	REUSABLE	(3.0)	(4.0)	3.0	.990	180D x 720	OAO	30000	400 NM; 35°
	REUSABLE	(3.0)	(4.0)	3.0	.990	180D x 720	SRS	5000	300 NM; 82°
SHUTTLE (72200 LB TO 100 NM; 28.5°)/ SPACE TUG	REUSABLE	(3.3)	(4.7)	3.7	.970	180D	MARS ORB.	9000	MARS INJECT
	REUSABLE	(3.3)	(4.7)	3.7	.970	180D	ERS	2100	SYNC. EQ. R.T.

*Average user's cost

Fig. 3-5 SELECTED LAUNCH VEHICLE/UPPER STAGE CAPABILITY/RELIABILITY/COST

1 nm = 1.852 km
1 lb = 0.4536 kg
1 in. = 0.0254 m

3.2 INITIAL BASELINE PAYLOAD DATA

This section summarizes the sources of the baseline payload data and provides overall and subsystem descriptions of the three initial baseline payloads, the Orbiting Astronomical Observatory (OAO-B), the Lunar Orbiter, and the Small Research Satellite (SRS). Included in this section are the apportionments of cost, weight and reliability for each payload by subsystem.

3.2.1 Payload Data Acquisition

Data on the baseline spacecraft, OAO-B, Lunar Orbiter and Small Research Satellite were obtained from sources as described below. After analysis of the data and allocation according to the standard study format, the data was resubmitted to the original source for review and comment. A summary of baseline payload and alternate derivatives is provided in Fig. 3-6.

OAO-B - The Goddard Space Flight Center (GSFC) through the Program Director of the OAO program provided LMSC with detailed historical cost and design data on the OAO program from its inception. Also, GSFC cooperated in interpretation of the detailed data. In addition, GSFC provided the results of the OAO/LST Shuttle Economics Study performed by the Grumman Aerospace Corporation.

Lunar-Orbiter - The Boeing Company provided all pertinent design and cost data generated during the conduct of the Lunar Orbiter program for NASA Langley Research Center (LaRC) under Contract NAS 1-3800. These included comprehensive design data, including 38 technical reports, specifications, plans, manuals, and flight test results. Program cost data were provided in the form of the Final Contractor Financial Management Report (NASA Form 533).

SRS - Design and cost data were provided to the study directly by the LMSC Program P-11 management and engineering personnel. Direct assistance on the technical details of the Small Research Satellite, its cost and schedule, were provided by the Chief Engineer of the program. As this is an LMSC developed

PAYLOAD	QTY. TO PERFORM MISSION OBJECT.	AVERAGE MISSION DURATION	APPROX. MISSION CHARACT. VEL. (FPS)	BASELINE LAUNCH VEHICLE	TERMINAL ORBIT	PAYLOAD WT. (LB) △3 SPACE-CRAFT	EXPERIMENTS	TOTAL	BASIC MISSION OBJECTIVE
OAO (B)	1	12 mos.	26,690	SLV-3C/Centaur	400 NM; 35° P=100 mins	3884 W_p=66	967	4811 W_p=66	Stellar UV Astronomy
LUNAR ORBITER (Ref. only)	5	Translunar + 30 days	35,920	SLV-3A/Agena	998 NM x 24 NM P=3.5 hrs	702 W_p=277	148	850 W_p=279	Hi-Resolution Apollo Sites + Photo Atlas
SYNC. EQ. ERS (Used in Parametric Analysis only)	4	12 mos.	33,650 (XFER) 39,500 (Total)	SLV-3A/Agena	19320 NM; 0° P=24 hrs	1435 W_p=884	226	1661 W_p=884	Hi-Resolution ER Imaging + Lo-Res. Phenomena
MARS ORBITER △2	5	218 days TransMars + 120 days	37,400	SLV-3C/Centaur	7330 NM x 110 NM P=8 hrs	1582 W_p=920	279	1861 W_p=920	Photo Atlas, Hi-Resol Sel. Areas + Secondary IR/UV
SRS	1	6 mos.	27,390	SLV-3A/Agena △1 Thorad/Agena	300 NM; 82° P=96 mins	202 W_p= 29	49	251 W_p=29	Ionospheric Measurements

△1 SRS launched as secondary payload on aft-rack of Agena.

△2 1971 Mars Mission

△3 Spacecraft and Total numbers include the propellant weight and expendables (N_2) weight; designated "W_p"

1 fps = 0.3048 m/sec
1 nm = 1.852 km
1 lb = 0.4356 kg

Fig. 3-6 BASELINE PAYLOAD MISSION CHARACTERISTICS

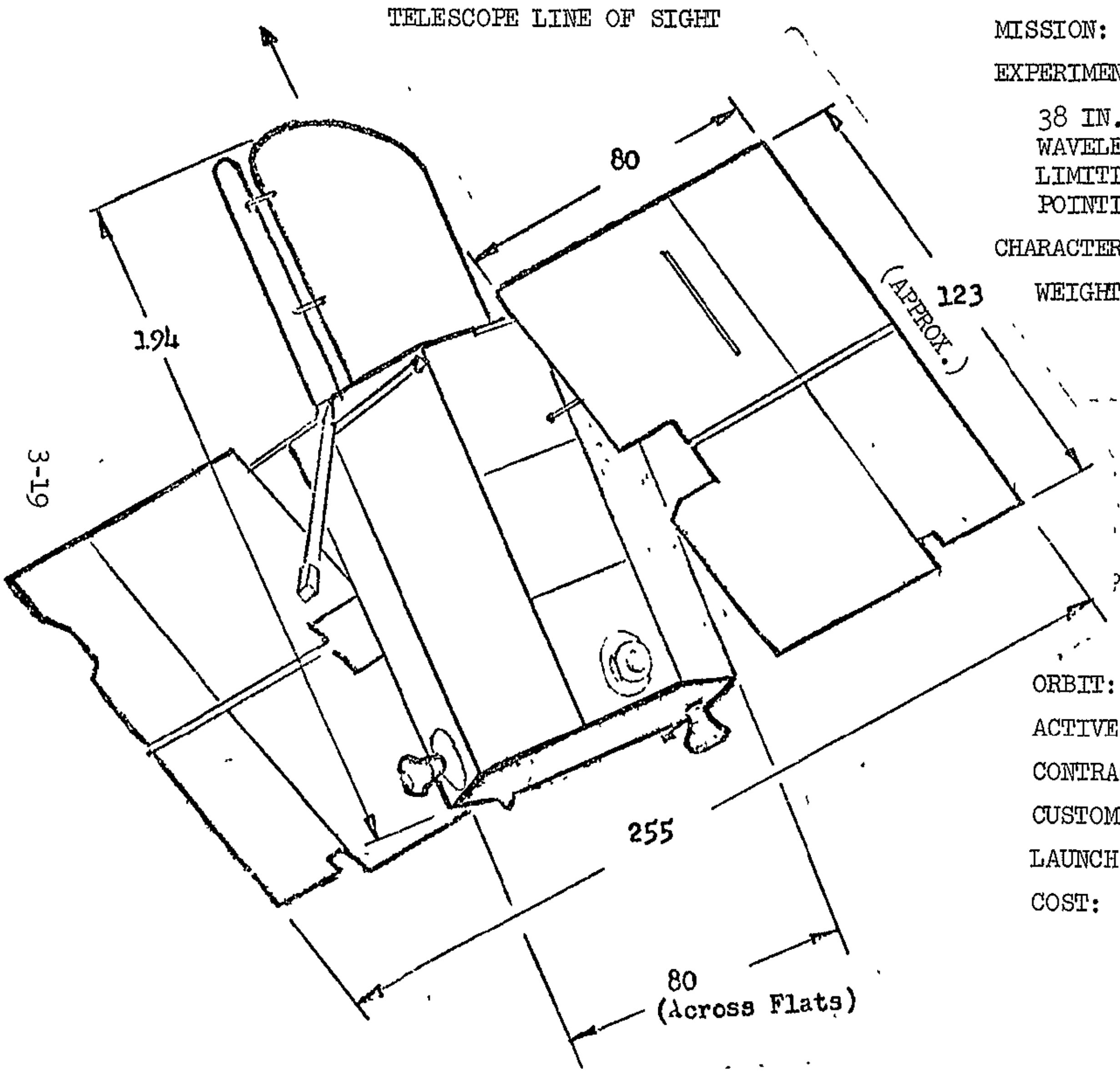
satellite, and several of the engineers that had participated in the SRS development are members of the study team, there were no problems related to acquisition or interpretation of the spacecraft details. Since most of the experiments flown during the 22 flights of the SRS were classified, a representative experiment package, suitably sized and priced, was selected for the SRS payload. This was the HIGLO experiment package, developed by LMSC for DoD for the OV-1 program. Concurrence on this substitution as a typical set of Space Physics experiments was obtained from Physics and Astronomy Programs, Office of Space Sciences and Applications. Descriptions of the HIGLO experiments are included.

3.2.2 Orbiting Astronomical Observatory (OAO-B) Baseline Data

3.2.2.1 General Description. The OAO is an unmanned astronomy satellite designed to gather scientific data from stellar sources from a circular earth orbit of 390 to 417 nm (718 to 768 km) with an inclination of $35^{\circ} \pm 1^{\circ}$. It is also capable of operating in an elliptical orbit, perigee 348 (641) and apogee of 520 nm (958 km). In agreement with NASA/Goddard, the OAO-B version of the series was selected as most representative of the stellar astronomical telescopes and most readily extrapolatable to the future large space telescopes of the 2-meter and 3-meter size.

The baseline OAO-B presents an octagonal cross-section body configuration with two large fixed (deployable) multi-panel rectangular solar arrays and with a protective sunbaffle protruding along the telescope line-of-sight approximately 76 in. (1.9 m) beyond the payload body length of 118 in. (3 m). The 38 in. (1 m) diameter telescope is mounted within a 48-in. (1.2 m) diameter cavity within the payload body. Figure 3-7 contains an illustration of the payload assembly and a listing of its primary characteristics.

3.2.2.2 Mission Description. The mission of the OAO-B comprises operation in earth orbit for a minimum of one year, pointing to a variety of stellar targets, and collecting and transmitting to earth high-resolution spectral data in the ultra-violet region of the spectrum.



MISSION: STELLAR UV ASTRONOMY

EXPERIMENTS:

38 IN. APERTURE CASSEGRAIN TELESCOPE FOR WAVELENGTHS 1100 Å TO 4000 Å
LIMITING STAR MAGNITUDE - 14
POINTING ACCURACY - 1 ARC SEC RMS

CHARACTERISTICS:

WEIGHT:	SPACECRAFT:	3884 LB
	EXPERIMENT:	967
	TOTAL	4811

DIMENSIONS:

BASIC SPACECRAFT:	80 IN. ACROSS FLATS x 118 INCHES
STOWED PACKAGE:	89 IN. x 91 IN. (OVER SOLAR ARRAYS) x 194 IN.
DEPLOYED PACKAGE:	255 IN. OVER SOLAR ARRAYS; 190 IN. OVER BOOMS

ORBIT: 400 NM x 35°

ACTIVE LIFETIME: 12 MONTHS

CONTRACTOR: GRUMMAN (SPACECRAFT)

CUSTOMER: NASA/GODDARD

LAUNCH VEHICLE: ATLAS/CENTAUR

COST: \$212 MILLION PROGRAM COST, INCLUDING \$36 MILLION FOR ONE FLIGHT ARTICLE

1 in. = 2.54 cm
1 nm = 1.852 km
1 lb = 0.4536 kg

Fig. 3-7 ORBITING ASTRONOMICAL OBSERVATORY

The sources and data are grouped as follows:

- Peculiar Stars - Time-dependent photometry of stars such as Beta Canis Majoris, T Tauri, and Wolf-Rayet.
- Normal Stars - Determination of energy distributed in the continuum, blanketing effects, and identification and intensities of strong emission lines.
- Nebular and Interstellar Media - Data on law of reddening, UV radiation field, and spectra of emission and reflection nebulae.
- Galaxies and Intergalactic Media - Data on spectral energy distribution of nearby galactic systems, and magnitude and intensity of Lyman-alpha red shift.

3.2.2.3 Subsystem Description. The subsystems of the OAO-B are:

Adapter - The baseline OAO-B is launched into orbit by an Atlas SLV-3C/Centaur. The OAO-B is mounted atop a Centaur/Payload Adapter; there are eight protruding fittings on the aft end of the payload, each of these mating with a forward ring on the adapter. A V-Band clamp ring secures the payload fittings to the adapter until pyrotechnic-actuated devices separate the clamp and allows separation of the OAO-B. A protective exit fairing covers the payload during launch and a portion of the ascent; the fairing is jettisoned prior to the aforementioned separation of the OAO-B from the Centaur. For purposes of this study, the exit fairing was considered part of the launch vehicle system; the adapter and separation clamp were considered part of the baseline payload system.

Experiments - The experiments comprise a telescope-spectrometer having the following characteristics:

Aperture:	38 in. (0.965 m)
Spectral Range:	1100 to 4267 Å
Resolution:	2, 8, 64 Å

Field Size:	5 min x 1 min 5 min x 10 sec 5 min x 40 sec
Pointing Accuracy:	1 sec
Data output:	digital

The experiment optical system employs a Cassegrain telescope with a large-aperture spectrometer. Principal elements are a primary mirror, a secondary mirror, spectrometer mirror, and diffraction grating. The experiment electronics system comprises an analog electronics assembly, seven detectors, and associated digital conversion electronics. Six UV detectors measure spectral energy and generate data in a train of asynchronous pulses (counted by a data accumulator in the digital electronics package). A seventh detector acquires data in the visible spectral range for correlating UV intensity and star magnitude.

Effective use of the experiment requires fine guidance within 1 to 2 secs of accuracy. The experiment package contains a fine-control error sensor which generates a star-presence signal indicating that a star has been acquired and allows transmission of error signals and (2) provides signals to ground on star presence, star magnitude, and error-monitoring signal.

Structures and Mechanisms - The baseline structure is an octagonal box, subdivided into 6 longitudinal segments and into 8 peripheral segments; a total of 48 equipment compartments surrounding a central 48 in. (1.2 m) diameter cylindrical cavity in which the telescope is installed. A separate angle-cutoff cylindrical sun shield is installed on the open telescope end. A "blow-off" lid covers the open end of the shield until jettison is commanded. An auxiliary semi-cylinder sunshield sun-protects the boresight star tracker.

Mechanisms include extendable balance booms and latches and extension mechanisms for the solar arrays.

Electrical Power Subsystem - The baseline subsystem consists of fixed-position arrays, nickel-cadmium batteries, and the power control, conversion, and distribution elements. A sun bathing capability is provided for orientation of the solar arrays to obtain high-energy input from the sun.

Guidance and Navigation (Stabilization and Control) Subsystem - The baseline subsystem stabilizes the OAO-B following separation from the Centaur, orients the payload to the attitudes required by the mission, and maintains the payload in precise pointing mode during the stellar-sighting periods. The GNS&C subsystem includes both coarse and fine-pointing equipment, the latter being switched on automatically or by ground command when the pointing error settles to within ± 2 min of arc. A boresight tracker, aligned parallel to the telescope optical axis, can be commanded to produce aiming offsets in increments of 15 sec up to ± 15 min of arc from the target star (5th magnitude or brighter) or in 1 min increments up to ± 1.5 deg from target star sighting line.

The principal elements of the S&C subsystem are: Three coarse and three fine momentum wheels and associated electronics; six star trackers; magnetic unloading equipment; an inertial reference unit; and integrating control electronics.

Attitude Control Subsystem - The baseline subsystem includes a primary and a secondary gaseous nitrogen system. Each consists of GN_2 tankage, plumbing and valves, and cold-gas thrusters. Control signals for on-off pulsing are provided by the GNS&C subsystem.

Communications, Data Processing, and Instrumentation (CDP&I) Subsystem - The baseline subsystem provides the following functions: (1) Ground command link for the spacecraft and the experiment package; (2) Telemetry ground links for wide and narrow band data transmission; (3) Tracking beacon for STADAN network; (4) Programming of spacecraft equipment, decoding and execution of spacecraft and experiment commands, and gathering and storage of spaceccaft and experiment status data; (5) Measurement of spacecraft status data. The frequencies used are:

Command	149.52 MHz
Wide Band Data	400 MHz
Narrow Band Data	136 MHz
Tracking Beacon	136 MHz

Multiple units are provided in redundant arrangement for the critical operating functions: four command receivers, two beacon transmitters, and two telemetry transmitters.

Environmental Control Subsystem - The baseline subsystem includes two basic types of hardware:

Passive Thermal Control

- Insulation
- Radiation coatings
- Conductive materials

Semi-Active Thermal Control

- Louvers
- Heaters
- Thermostats

The louvers are used in the high-heat-producing equipment compartments.

3.2.2.4 Initial OAO-B Baseline Cost Estimate. The source for OAO-B cost data was the NASA Goddard Space Flight Center (GSFC) at Greenbelt, Maryland. Since the cost data were not in a pure form for this particular spacecraft, a methodology for isolating the OAO-B costs from the OAO series was established and agreed upon by the LMSC and the Goddard cost analysts.

To establish a reasonable OAO-B spacecraft repeat-flight cost, the repeat cost quoted by GSFC for OAO-C was used, less any one-time costs for the 'C' configuration (including an LMSC estimate of one-time thermal analysis). Since the Goddard subsystem breakdown differed slightly from the study subsystem format, some costs were reallocated utilizing the component backup cost data supplied by Goddard.

In addition, the cost of the OAO-B experiment package was allocated among one-time costs (including prototype), repeat unit costs, and operations costs (primarily data reduction and analysis). This allocation was based on ratios derived from LMSC interpretation of the Smithsonian Astrophysical Observatory (SAO), Wisconsin Experiment Package (WEP), and Princeton experiment cost breakdowns.

To approximate the magnitude of OAO spacecraft research and development needed to attain an OAO-B level of capability, the GSFC-supplied total spacecraft costs through OAO-A-2 were added to the one-time costs for the B configuration (quoted OAO-B cost less the repeat cost derived above), and then reduced by the estimated costs of OAO-A-1 and A-2 flight articles and adjusted to a 1970 dollar base. Costs through OAO-A-1 were adjusted from a 1962-66 base, costs for A-2 from a 1967-68 base, and costs for B from a 1969 base.

The operations costs derived by LMSC include launch operations and services (functional tests, mating, etc.), flight operations and services (OCC team operation and software support), and tracking/data acquisition net operations.

The OAO-B costs were subjected to a further allocation of costs to individual subsystems. This step consisted of prorating certain of the costs that appeared amenable to further allocation, particularly Spacecraft Integration and Test, GOE/GSE, and Operations costs. In making these allocations best judgment was used as to the relative magnitude of these cost elements in relation to the individual subsystems. The results of this allocation, adjusted to 1970 dollars, are reflected in the final OAO-B Baseline Cost Summary, Fig. 3-8.

All costs were reassessed subsequently and a bottom-up cost estimate for the program was made. Relatively good correlation was found. The costs for the "Recosted Baseline" are shown in Section 6.3.

PAYLOAD: OAO-B BASELINE MISSION: EARTH ORBIT

($ IN THOUSANDS) - 1970 $

COST CATEGORY / SUBSYSTEM		NON - RECURRING COSTS					RECURRING COSTS					TOTAL PROGRAM COST
							HARDWARE			OPERATIONS		
		DEVEL.	GSE	S/C INT. & TEST	PROC. MGM'T.	TOTAL	QTY	AVE. UNIT	TOTAL	AVE. UNIT	TOTAL	
PAYLOAD ASSY. AND INTEGRATION		$ 500	$ 100	-	-	$ 600	1	$ 150		$ 100		$ 850
EXPERIMENTS AND MISSION PECULIAR EQUIPMENT		8717	-	-	-	8717	1	7800		2200		18,717
STRUCTURES AND MECHANISMS		5356	438	$ 3250	-	9044	1	5100		20		14,164
ELECTRICAL AND PYROTECHNICS		14568	730	1785	-	17083	1	2900		550		20,533
GUIDANCE, NAVIGATION, STABIL., AND CONTROL		65849	6520	6100	-	78469	1	11700		3900		94,069
PROPULSION AND ATTITUDE CONTROL		2245	730	300	-	3275	1	300		200		3,775
TELEMETRY, TRACKING, AND COMMAND		33063	4460	3300	-	40823	1	4600		2800		48,223
ENVIRONMENTAL CONTROL		5299	146	600	-	6045	1	1000		550		7,595
SUB TOTAL	ALLOCATED	135597	13124	15335	-	164056	1	33550		10320		207,926
	NON-ALLOCATED TO SUBSYSTEM	500			$853	1353	1	2600*		900		4,853
PAYLOAD TOTAL		136097	13124	15335	$853	165409	1	36150		11220		212,779

* Program Management

Fig. 3-8 Initial Program Cost Apportionment - OAO-B

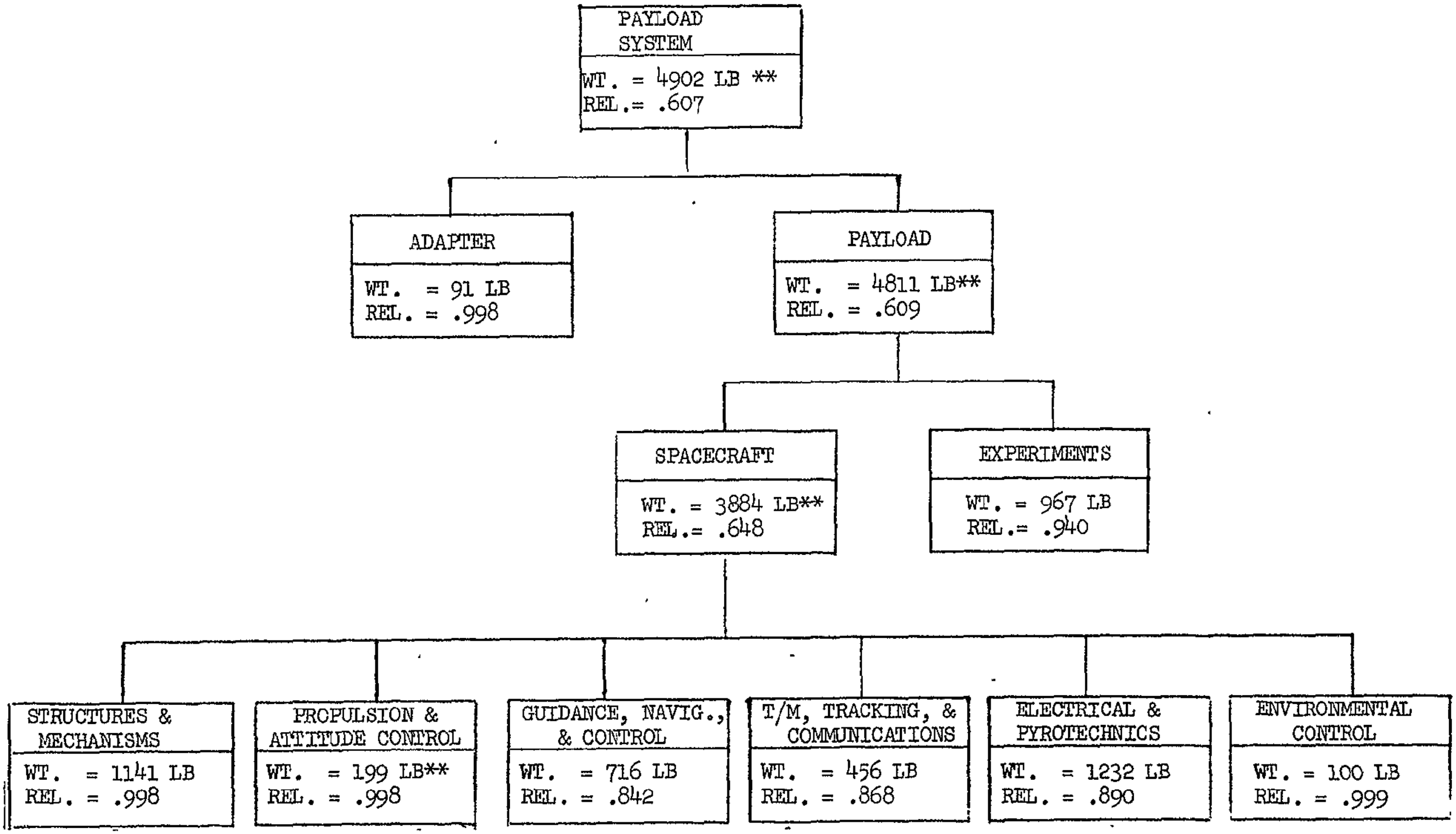
3.2.2.5 OAO Weight and Reliability. The summary of subsystem, payload, and payload system weights and reliabilities for the baseline OAO-B are tabulated in Fig. 3-9. The 4,811 lb (2,187 kg) (including 66 lb [30 kg] of expendable N_2) is the total weight of the payload as it is placed into orbit by the Centaur upper stage. The 0.609 reliability is the probability that the OAO-B will perform its mission for one year in orbit without catastrophic failure.

The reliability data on the OAO-B were not available from NASA/Goddard in the data package provided to LMSC. Using the weight statement on the OAO-B and historical data on similar hardware, LMSC created a detailed listing of weights and estimated reliabilities for the components of the subsystems. This list was coordinated with NASA/Goddard and their comments have been incorporated.

3.2.3 Lunar Orbiter Baseline Data

3.2.3.1 General Description. The Lunar spacecraft was developed for NASA Langley Research Center by the Boeing Company under Contract NAS 1-3800. The program initiated in 1964, had the objectives of providing high-resolution photography (< 1 meter) of potential Apollo landing sites and a medium resolution photo atlas of the moon. A total of five successful spacecraft missions in five attempts were completed during 1966 and 1967; all program objectives were accomplished. The Lunar Orbiter is of particular interest in examining minimum cost design approaches as the design was severely constrained in weight by the capability of the launch vehicle and by direction to utilize available systems and components in order to expedite the development phase. Weight restrictions forced costs upwards by dictating such weight reduction techniques as use of chem-milled titanium tanks, the extensive use of beryllium and a total redesign of an existing photographic subsystem.

The Lunar Orbiter spacecraft weighed 850 lb (386 kg), including expendable propellants, when separated from its launch vehicle, the Atlas SLV-3 A/Agena. The following paragraphs describe the baseline Lunar Orbiter spacecraft subsystems, and the general configuration of Lunar Orbiter and spacecraft features are shown in Fig. 3-10.

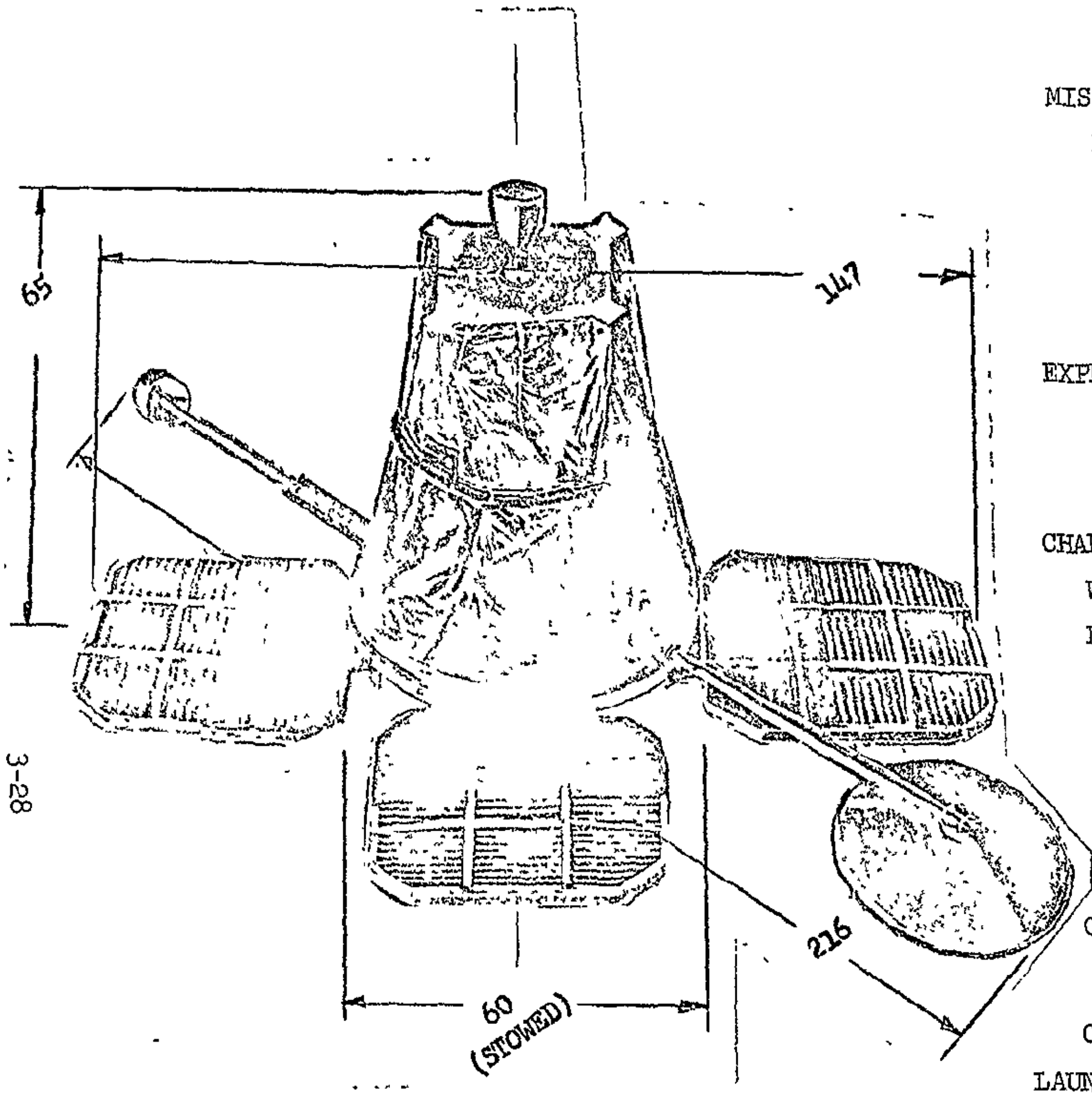


* Hardware comprises V-band Separation Clamp (16 lb) and OAO/Centaur Adapter Cone (75 lb). Reliability product includes these functional elements plus the exit fairing jettison function.

** Includes 66 lb of N_2 expendable.

1 lb = 0.4536 kg

Fig. 3-9 Reliability & Weight Apportionment - Baseline OAO-B



MISSION:

PRIMARY - HIGH-RESOLUTION PHOTOGRAPHY OF APOLLO LANDING SITES

SECONDARY - PROVIDE PHOTOGRAPHS FOR LUNAR ATLAS

- MEASURE MOON'S GRAVITATIONAL FIELDS

EXPERIMENTS: HIGH-RESOLUTION CAMERA

MEDIUM-RESOLUTION WIDE-ANGLE CAMERA

ON-BOARD FILM PROCESSOR

CHARACTERISTICS:

WEIGHT: 850 LB

DIMENSIONS: 5 FT DIA. x 5.5 FT LONG (STOWED)
18 FT OVER ANTENNAS
12.2 FT OVER EXTENDED SOLAR ARRAYS

INITIAL ORBIT: 110 x 1000 NM LOWERING TO 25 NM PERILUNE (3.5 HR PERIOD)

ACTIVE LIFETIME: 30 DAYS

CONTRACTOR: BOEING COMPANY (ASSOCIATES AND SUBS: KODAK, RCA, GENERAL DYNAMICS, LMSC)

CUSTOMER: NASA/LANGLEY

LAUNCH VEHICLE: ATLAS (SLV-3A)/AGENA

COST: $150 MILLION (APPROX.)
SPACECRAFT & GSE FOR 5 FLIGHT ARTICLES + 3 GROUND TEST ART.

1 lb = 0.4536 kg
1 nm = 1.852 km

Fig. 3-10 LUNAR ORBITER SPACECRAFT

3.2.3.2 Mission Description - The primary objective of the Lunar Orbiter mission was to obtain high resolution photography of the prospective Apollo manned lunar landing sites. The secondary mission objective was to provide photographic data of the entire lunar surface to be used in preparing a detailed lunar atlas. A description of the lunar orbit mission sequence is as follows:

Following injection into translunar orbit and launch vehicle separation, the spacecraft is oriented to the sun and solar arrays and antennas are extended. After DSIF measurement of injection errors, a midcourse maneuver is commanded and executed. Upon lunar arrival, the spacecraft is commanded to reorient and brake into an initial orbit with apolune altitude of 998 nm (1840 km) and eccentricity 0.287. After tracking and determining the orbit elements, perilune is lowered to 24 nm (4.4 km) by propulsive braking and the final lunar orbit with a period of 3.47 hrs and eccentricity of 0.336 is attained. During coast operation the spacecraft is oriented with the roll axis pointed at the sun for maximum solar incidence angle. For photo operations, the spacecraft is reoriented with the camera axis pointed at the area of interest. A V/H sensor measures drift and commands IMC and yaw. Photography commences upon command. The medium resolution camera photographs an area of 31.6 by 37.4 km and, simultaneously, the high resolution camera photographs an area of 16.6 by 4.15 km with the same center point. The exposed film passes through a processor dryer where it meets Bimat web that provides development and fixing chemicals. The film is dried and stored on a readout looper. Processing time is about 5 min per complete frame. Film passes next through the readout scanner and is stored on the takeup reel. After completion of all photography, the Bimat web is cut and the film is rerun through the scanner for readout.

3.2.3.3 Subsystem Descriptions. The principal subsystems comprising the Lunar Orbit baseline payload are as follows:

Adapter - The spacecraft lower (equipment mounting) deck is attached to the launch vehicle payload adapter by a V-band clamp. During ascent, the payload is protected by an aerodynamic fairing. Upon final burnout, the Agena commands pyrotechnics to fire and open the V-band clamp. Separation is effected by springs.

Experiments - The primary experiment is the Eastman-Kodak Photographic payload. This camera system, developed from an earlier operational system, is housed in a pressurized, temperature controlled housing. The system includes a dual lens camera with high and medium resolution lenses, image motion compensation controlled by a V/H sensor (which also controls spacecraft yaw during photography), a Bimat processor dryer, an optical-mechanical scanner with video readout, Bimat and film, and film transport. The system is capable of imaging a total of 194 frames of both high (1 m) and medium resolution photography. Secondary experiments include micrometeorite detectors and radiation monitors.

Structure and Mechanisms - The structure consists of the equipment mounting deck, tank deck and engine decks supported by trusses and arches. During ascent, solar arrays and antenna booms are folded: following separation, they are deployed.

Electrical Power Subsystem - Primary power is provided by four solar panels of of 13.1 ft^2 (1.2 m^2) each which provide 87.5 watts each. A secondary NiCad battery provides power during solar occultation. Regulators and controllers protect the spacecraft and the electrical subsystem from fluctuations, and the battery from overcharge.

Guidance and Navigation (Stabilization and Control) Subsystem - Spacecraft attitude reference is provided by a Canopus star tracker and sun sensors. An inertial reference unit maintains control when these references are furnished to the Attitude Control and Propulsion subsystems. The S&C subsystem includes the Flight Control Electronics Unit which sets deadband gains, conditions and routes signals. The subsystem includes a digital programmer which stores precommanded sequences and operations which are executed by time. Commands include attitude changes, velocity changes, camera operation and data readout.

Propulsion and Attitude Control - Velocity control is provided by a 100 lbf. (445 newtons thrust) liquid bipropellant (N_2O_4/Aerozine 50), hypergolicly ignited, pressure fed engine. Propellants are stored in tanks mounted below the engine and are provided to the engine upon command by N_2 pressurized bladders.

Valves and regulators are provided to control pressurant flow and isolate the system from the attitude control subsystem which uses the same N_2 supply. During engine firing, pitch and yaw control are provided by gimballing the engine. The system is capable of providing a total of 978 m/sec for midcourse correction, lunar orbit injection and orbit adjustment. The spacecraft is three-axis stabilized by N_2 reaction thrusters during coast and orbit operations. The attitude control subsystem also provides roll control during engine firing. An N_2 tank stores 14.7 lb (6.7 kg) of dry N_2 which is regulated to 20 psi (13.790 newtons/m^2) and is provided to the thrusters upon command of the G&N subsystem.

Communications, Data Processing, and Instrumentation Subsystem - The CDP&I subsystem consists of two major packages. Package A, developed by RCA, includes the high gain antenna for video data transmission, an antenna pointing control unit, a modulation selector which selects the data mode, a command decoder and transponder for providing tracking data and receiving commands from earth, and a 10 watt traveling wave tube amplifier for amplifying video data for transmission.

Package B includes a low gain antenna for status data transmission, a PCM telemetry multiplexer/encoder, signal conditioners and transducers.

Environmental Control Subsystem - The spacecraft is enclosed in a mylar thermal barrier with a port for the Canopus Star Tracker and a camera thermal door. The engine deck is isolated from the spacecraft interior by an insulated heat shield. The Equipment Mounting Deck (EMD), which is normally oriented towards the sun, has a high emittance coating. Tank heaters are provided. The Photographic Subsystem has its own environmental control and radiates excess heat to the EMD radiating surface.

A major cost element of the Lunar Orbiter program was for the ground operations equipment and support. An extensive ground data receiving and recording system was installed in the NASA Space Flight Operations Center (SFOF) and the Deep Space Instrumentation Facility (DSIF) for handling received photo data and for commanding and controlling the spacecraft.

3.2.3.4 Lunar Orbiter Cost Data - The Lunar Orbiter historical cost data were compiled from NASA #533 forms obtained from The Boeing Company, which included the complete costs for the Boeing portion of the Lunar Orbiter effort as well as the Eastman Kodak subcontract for the photo subsystem and the RCA work in the power and CDPI areas. The Lockheed costs associated with the Lunar Orbiter adapters, which were GFE to Boeing, were obtained from the LMSC accounting records.

The NASA #533 form data does not provide a split of non-recurring and recurring costs; costs are accumulated under categories of engineering, production, developmental, etc., labor, material, purchased parts, and overhead type accounts. The split of RDT&E and unit hardware costs was performed by LMSC. The data provided a cost breakdown by major components, subsystems, and support categories, and the contractor responsible for the portion of work.

The apportionment of Eastman Kodak (EKC) effort was guided by percentages of non-recurring and recurring costs by major cost category such as payload hardware, GSE, STE & Tooling, and Program Management supplied by Eastman Kodak. EKC also advised on the allocation of operating costs and elimination from the totals of most of the additional $2.0M EKC subcontract, which was primarily for additional copies of the lunar surface photographs and not part of the original mission costs. The operating costs for this subsystem are quite high due to the ground support effort required in terms of equipment lease and personnel for the photo transmission operations.

The Boeing and RCA Lunar Orbiter costs were apportioned into recurring and non-recurring categories using in-house subsystem CERs as a guide to expected total subsystem cost magnitude. The actual apportionment of non-recurring and recurring costs was done by subsystem and component by allocating such categories as developmental effort and tooling to RDT&E. The purchased parts and production costs were charged to recurring cost. The engineering and QA were split using judgment and knowledge of the type of effort involved in the particular subsystem development or procurement. The overhead accounts were allocated in approximate proportion to the non-recurring/recurring split obtained from above allocations.

The subcontractors fees were included in the Lunar Orbiter baseline costs, but the Boeing and Lockheed fees are excluded to maintain consistency with other payload data in this study.

The apportioned major component costs were then summarized into subsystems in accordance with the Payload Effects Study subsystem breakdown. The remaining non-subsystem associated non-recurring costs such as test program, sustaining engineering, and integration costs were prorated to each subsystem based on the magnitude of cost in the subsystem and tabulated under the S/C Integration and Test category in Fig. 3-11. This category also includes costs already identified by subsystem in addition to the prorated costs. The GSE cost was identified in all but the Guidance, Propulsion, and Structures subsystems. Thus, these include prorated GSE costs. The Program Management category includes identifiable costs from the subcontractor's, as well as prorated Boeing program management by subsystem.

The RDT&E unallocated costs consist of mission computer programs, launch operations and flight operations planning, and miscellaneous small categories which were not allocated by subsystem.

The recurring hardware costs include allocated program management cost on basis of subsystem hardware cost.

The operations costs were built up by subsystem from identifiable subcontractor's cost categories and the allocation of logistics, launch operations, flight operations, program management, and sustaining engineering on basis of subsystem unit hardware costs. The unallocated operations costs include mission computer programs and miscellaneous support costs which were not allocated.

The apportioned Lunar Orbiter costs were checked for completeness and converted to the 1970 dollar base. The apportioned Lunar Orbiter cost data in 1970 dollars are shown in Fig. 3-11.

PAYLOAD: LUNAR ORBITER - BASELINE (REFERENCE)   MISSION: LUNAR ORBIT

($ IN THOUSANDS) - 1970 $

COST CATEGORY / SUBSYSTEM	NON-RECURRING COSTS					RECURRING COSTS					TOTAL PROGRAM COST
						HARDWARE			OPERATIONS		
	DEVEL.	GSE	S/C INT. & TEST	PROG. MGMT.	TOTAL	QTY	AVE. UNIT	TOTAL	AVE. UNIT	TOTAL	
PAYLOAD ASSY. AND INTEGRATION (1)	4661.7	-	939.1	145.6	5746.4	5	480.5	2402.8	540.6	2702.8	10,852.0
EXPERIMENTS & MISSION PECULIAR EQUIPMENT	24979.3	7451.1	4148.6	2139.9	38718.9	5	2048.3	10241.3	1460.3	7301.8	56,262.0
STRUCTURES AND MECHANISMS (2)	2578.9	921.7	2087.7	207.9	5796.2	5	1318.0	6590.0	294.9	1474.5	13,860.7
ELECTRICAL AND PYROTECHNICS	4314.2	1708.4	2039.9	1539.1	9601.6	5	1262.6	6313.1	613.0	3065.1	18,979.8
GUIDANCE, NAVIGATION, STABILIZATION & CONT.	3605.9	4409.2	4290.4	665.7	12971.2	5	2393.2	11966.1	872.7	4363.3	29,300.6
PROPULSION	2148.5	1634.0	2034.0	314.6	6131.1	5	789.0	3945.0	289.6	1447.8	11,523.9
ATTITUDE CONTROL	560.5	408.4	508.5	79.8	1557.2	5	115.4	576.9	42.5	212.5	2,346.6
TELEMETRY, TRACKING AND COMMAND	8191.8	9752.8	2767.9	1642.3	22354.8	5	2151.3	10756.7	965.1	4825.7	37,937.2
ENVIRONMENTAL CONTROL	55.0	-	45.0	5.0	105.0	5	16.5	82.5	5.0	25.0	212.5
SUB-TOTALS: ALLOCATED	51095.8	26285.6	18861.1	6739.9	102982.4	5	10574.8	52874.4	5083.7	25418.5	181,275.3
SUB-TOTALS: NON-ALLOCATED TO SUBSYSTEM	4987.7	-	-	-	4987.7	5	203.6	1017.6	670.1	3350.6	9,355.9
PAYLOAD TOTAL (3)	56083.5	26285.6	18861.1	6739.9	107970.1	5	10778.4	53892.0	5753.8	28769.1	190,631.2

(1) INCLUDES COSTS OF SHROUD & ADAPTER
(2) INCLUDES SPACECRAFT ASSY. & INTEGRATION COSTS
(3) EXCLUDES FEE AND NASA PROGRAM MANAGEMENT

Fig. 3-11   PROGRAM COST APPORTIONMENT - LUNAR ORBITER (LUNAR ORBIT MISSION)

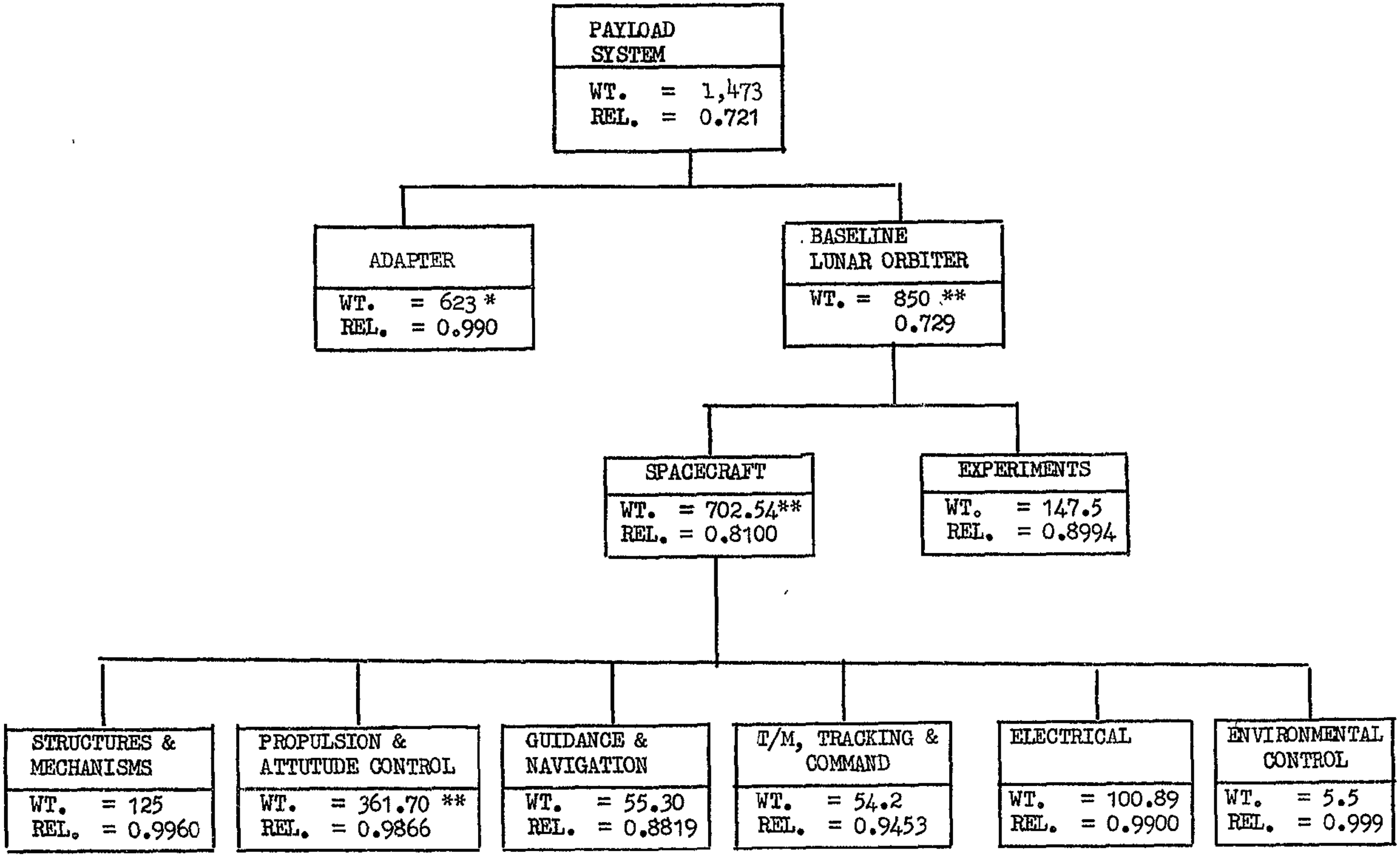
3.2.3.5 Lunar Orbiter Weight and Reliability Data. The Boeing Company reli bility numbers for the principal components and subsystems of the Lunar Orbi were carefully analyzed and are summarized in Fig. 3-12.

3.2.4 Small Research Satellite (SRS) Baseline Data

3.2.4.1 General Description. The Small Research Satellite (SRS) was develc by LMSC as a subsatellite to be orbitally launched by the Agena spacecraft. The SRS was created during several classified contracts to satisfy a need of secondary experimenters whose requirements were limited by those imposed by the primary host carrier spacecraft. As a separable autonomous subsatellite it can operate totally independent of the host offering the experimenter com plete freedom of operation. The sole restrictions that are imposed upon the spacecraft by the host vehicle are that it must constrain itself to the weig and volume envelope provided by the host and that it will not interfere with or endanger the primary carrier. In size, weight, capability, cost, and com plexity it is quite similar to several small NASA satellites such as Explore and Pioneer and its primary experiments have fallen into the generic classi- fication of Space Physics and Applications. The maximum weight permissible including optional equipment, experiments and launch vehicle interface equi ment is 400 lbs (182 kg). Designs for using SRS as the primary dedicated pa load for the Delta and Scout launch vehicles have been provided by LMSC to I The baseline spacecraft is shown in Fig. 3-13.

3.2.4.2 Mission Description. The 22 SRS flights to date have been in polar orbit. For the baseline mission a polar orbit at 300 nm (552 km) was selec This orbit is consistent with the requirements of the selected Space Physic experiment (HIGLO) and with the nominal requirement of 6 month's flight dura tion. A typical program of 2 years' continuous observation is planned.

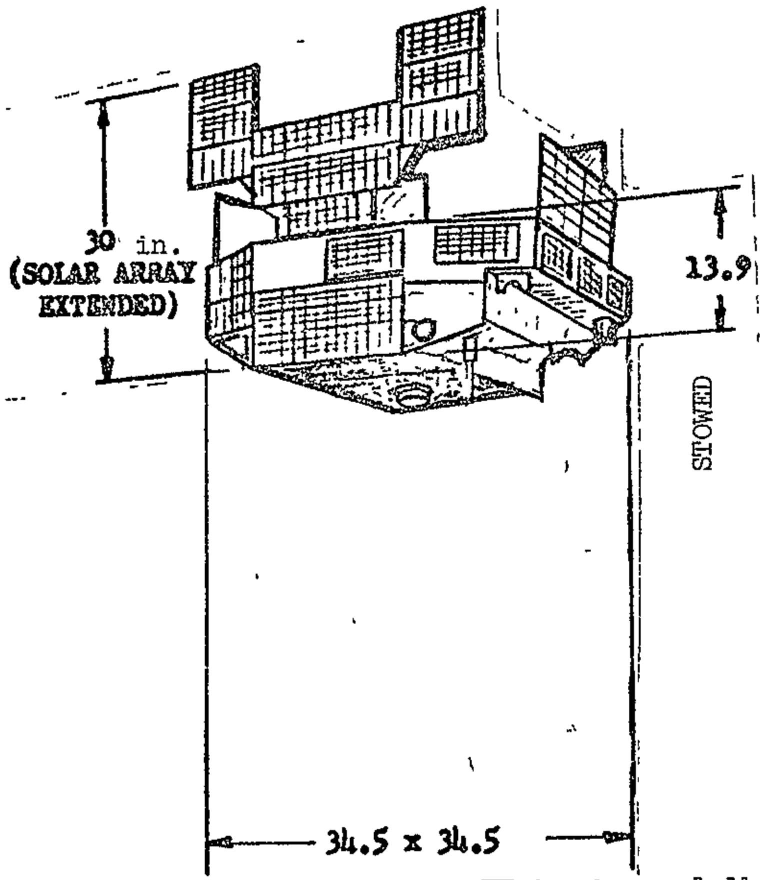
3.2.4.3 Subsystem Description. The SRS subsystems for the accomplishment of the HIGLO mission are summarized below:



* Includes Adapter which remains with the injection stage.
** Includes Propellants, Residuals and Gases equal to 277.43 lbs.

1 lb - 0.4536 kg

Fig. 3-12 Baseline Lunar Orbiter (Lunar Orbit Mission) - Reliability & Weight Apportionment



MISSION: SUB-SATELLITE LAUNCHED FROM AGENA SPACECRAFT INTO AN INDEPENDENT ORBIT, USING INTEGRAL PROPULSION.

BASIC SPACE RESEARCH EXPERIMENT CARRIER. ALTERNATE APPLICATION AS ENGINEERING EXPERIMENT CARRIER FOR COMMUNICATIONS AND PROPULSION EXPERIMENTS.

EXPERIMENTS: HIGLO, see Fig. 3-14

CHARACTERISTICS:

WEIGHT: 202 LB BASIC
49 LB EXPERIMENTS (VARIABLE WITH MISSION UP TO 100 LB.)
251 TYPICAL TOTAL

DIMENSIONS: 34.5 x 34.5 x 13.9 BASIC ENVELOPE (STOWED)(SOLAR ARRAYS & EXTENDABLE BOOMS INCREASE THE 13.9 DIMENSION)

ACTIVE LIFETIME: 1.5 YRS + WITH VARIOUS ORBITS

CONTRACTOR: LOCKHEED MISSILES & SPACE COMPANY

COST: APPROX. AVG. UNIT COST $1.3 MILLION (EXCLUDING EXPERIMENTS)

LAUNCH VEHICLE: ATLAS SLV-3A OR THORAD WITH AGENA PRIMARY ALTERNATES INCLUDE DELTA AND SCOUT

1 lb = 0.4536 kg
1 in = 0.0254 m

Fig. 3-13 SMALL RESEARCH SATELLITE (LMSC P-11 SUB-SATELLITE)

Adapter - The SRS system for subsatellite launch consists of a shear panel which is mounted on the aft rack of the Agena spacecraft. The shear panel includes all electrical interfaces with the host. Prior to launch, the shear panel is extended 90° to the longitudinal axis of the host while the host maintains attitude stability. The SRS separates, spins up and the transfer rocket fires. In the dedicated mode, the SRS is mounted directly atop the launch vehicle and is separated following terminal stage firing.

Experiments - Due to the fact that all P-11 experiments that had actually flown were of a classified nature it was necessary to define a set of suitable and representative experiments for the baseline SRS payload for use during the study.

An existing experiment package HIGLO which flew on USAF OV-1-18 fulfilled all of the above requirements and since it was developed at LMSC its design and cost data were readily available. This package was selected as the baseline for the SRS. The HIGLO consisted of 12 sensors and supporting equipment. Changes from OV to SRS would include addition of a 3-axis fluxgate magnetometer and the provision of attitude reference data by an earth sensor. The earth sensor is a unit of the basic SRS equipment and is needed to attitude reference the sensor data. The individual sensors and components of HIGLO are shown in Fig. 3-14.

Structures and Mechanisms - The SRS has a simple three-bay structure which has been contoured to fit the aft rack of the Agena with adequate clearance between itself and the interstage to the lower propulsive stage. The center bay contains the solid rocket propulsion in the middle and basic spacecraft units such as TT&C and power subsystem components in both ends. The outer bays (wings) are reserved for mounting experiments and for supporting the solar panels. The total volume available in the wings for experiments is 3.5 ft^3 (0.099 m^3).

Electrical - Primary power for SRS is provided by extendable solar arrays. A secondary NiCad battery provides power during nighttime and supplements the array power during peak power demand periods. The nominal system is capable

ARPA Experiment No. 819 HIGLO Investigation of Horizontal Ion Density Gradients in Lower Atmosphere
Principal Investigators: G. W. Sharp and R. G. Johnson, LMSC
Flight Date: 3/17/69, H_p = 362 nm, Period = 95 mins, Incl. 98.8°

Experiment Summary

Exp. #	Experiment Name	Wt(lbs)	Pwr.Avg.	Dim.	Vol(in^3)	Remarks
-1	Ion Energy Analyzer (4)	4.7 lbs	5.2	5x5x3	75	4 sensors
-2	Epithermal Electron Analyzers (3)	5.5 lbs	4.8	6.2x4.9 x4	121.5	3 sensors
-3	Cyl. Langmuir Probe	2.0 lbs	3	7x4.5x2	63	incl 15" probe
-4	Electrostatic Analyzer	5.9 lbs	2.3	6x7.5x7	315	3 sensors at diff. angles
-5	Multichannel Particle Analyzer	2.6 lbs	1	6x3.3x5	99	
-6	Multichannel Particle Analyzer	2.8 lbs	1	6x3.3x5	99	
-7	Multichannel Particle Analyzer	2.2 lbs	1	6x3.3x5	99	
-8	Proton Hydrogen Analyzer	2.9 lbs	0.9	6x4x6	144	
-9	Total Energy Proton Sensors (2)	3.3 lbs	0.8-1.4	(2) 10x3	60	2 sensors
-10	Angular Distribution Instruments (3) plus power supply	6.3 lbs	3.2	3x3x11 (3) 3x2x5	129	3 sensors and pwr. supp.
-11	Penetrating Radiation Monitor	5.9 lbs	2.6	4x2x4.5	36	
-13	Electric Field Probe	2.6 lbs	1.3	2x4x5.5	44	
-15	3 Axis Magnetometer (Flux gate)	0.6 lbs	1	3"x1" dia.	9 in	0.01 gauss resolution
-12	Calibration & Interface Box	1 lb	0.3	3x3x3.5	31.5	support equipment
-14	Data Mode Box	1 lb	1.1	4x2.6x1.6	16.7	
		49.3 lbs	30.1 watts		1341.7 in^3	

Other data: 196 prime meas. at 1 sps (196 bps) or 40 sps (20 kbps) 11 subcomm at 1/64 sps, 13 discrete cmds 2 hr. readin capacity on tape recorder 7.5 min readout. Desired Op Temp Exp 1&2 70°F, 3-13 59-85°F. Need to know pointing direction with respect to velocity vectors.

1 nm = 1.854 km
1 lb = 0.4536 kg
1 in. = 0.0254 m

Fig. 3-14 EXPERIMENT DATA SHEET

of providing 500 w-hr/day (1,800,000 J/day) to the entire payload. The spacecraft requires 156 w-hrs (562,000 J/day) leaving 344 w-hrs (1,238,000 J/day) available to the experiments.

Guidance, Navigation and Stabilization - The baseline SRS is passively spin-stabilized at 60 to 85 RPM. Initial spin is provided by two small solid rockets. A nutation damper in the form of a mercury tube, is provided. Attitude reference is provided by an infra-red earth sensor. Alternate stabilization systems including active control spin stabilization and three axis gravity gradient or wheel stabilization have been designed for SRS, however for this study only the passive system was considered.

Propulsion and Attitude Control - The SRS can be configured with several different combinations of solid rockets for propelling it to various secondary orbits. In the baseline, two rockets are provided for initial orbit transfer and for circularization into the final orbit.

Command, Data Processing, and Instrumentation - As with the other subsystems, SRS' CDPI system is available in a number of options dependent upon user requirements. The baseline system consists of VHF FM/FM telemetry, onboard tape recorders and a tone-digital command system. A primary timer provides for system sequencing. Alternate configurations include UHF S-Band Telemetry, PCM Telemetry and digital command systems with various antenna configurations.

Environmental Control - The SRS is passively thermally controlled through use of thermal shields, optical solar reflectors and paint. Supplemental control, such as heaters can be provided if required.

3.2.4.4 SRS Cost Data. Recurring costs for the SRS were derived by recasting existing cost data on the Lockheed P-11 subsatellite into a format compatible with this study, using a bottom-up cost estimating methodology. Historical data on the calendar year 1968 unit costs of a P-11 configuration similar to the SRS were used as the basis for this bottom-up estimate. These data, provided by the P-11 Project Office, comprised the following:

- Material, subcontract and manpower expenditures for hardware down to the component and major assembly levels.

- Manpower expenditures for assembly, test, integration, sustaining engineering, and program management at the subsystem and system levels.

- Manpower expenditures for the launch and mission operations phases of the program.

Using these cost inputs, the recurring-cost estimate was generated by applying the given numbers against a Work Breakdown Structure tailored to the SRS configuration. To normalize these costs from the 1968 levels supplied by the Project Office to 1970 values, the appropriate aerospace-industry inflation factors were applied.

With respect to nonrecurring costs for the SRS, no appropriate historical data were available because of security restrictions imposed by the using programs, and also because of the incremental step-function type development in which the SRS has evolved. Therefore it was necessary to formulate a representative SRS nonrecurring cost by analysis. This was done by applying parametric costing techniques to the technical characteristics of the SRS configuration. Unmanned-spacecraft Cost Estimating Relationships were used with SRS subsystems definitions to generate subsystem level nonrecurring costs; these were then summed to obtain an overall SRS nonrecurring cost estimate. Particular judgment was exercised in applying the CERs to account for the peculiarities of the P-11 program, such as minimum documentation and experimental-shop manufacturing procedures. The final data resulting from this procedure were approved by the P-11 Program Office as representative of the SRS non-recurring cost. The HIGLO experiments were costed using actual cost data.

Finally, to achieve a greater distribution of costs to the subsystems, certain of the costs accrued at system level, such as program management and systems engineering, were prorated against subsystems. These allocations were made on

a best judgment basis, taking into account the relative weighting of these functions for each subsystem. The baseline costs of the SRS vehicle derived in this way are listed in Fig. 3-15.

3.2.4.5 SRS Weight and Reliability Estimates. The summary of subsystem, payload, and payload system weights and reliabilities on the SRS are tabulated in Fig. 3-16. The hardware represented in the Adapter is the Shear Panel Assembly which acts as a launch platform for the baseline SRS (mounted on the aft equipment rack of an Agena upper stage). The 250.8 lb (113.8 kg) and a reliability of 0.556 represents the payload as it is separated into its interim orbit.

3.3 ANALYSIS REVISION AND UPDATE OF BASELINE PAYLOAD DATA

Following the initial selection of the OAO-B, Lunar Orbiter and SRS as baseline payloads, reexamination of the payloads with the view of correlating to the NASA mission model led to the decision to synthesize certain specific derivatives of the baselines. Within the traffic model, there were no planned Lunar Orbiter missions per se; however, there were many synchronous equatorial missions. Furthermore, although there was not a sizable amount of planetary traffic, the cost of planetary exploration represented a considerable portion of the unmanned funding requirements. Thus, it was decided to synthesize derivatives of the Lunar Orbiter for these two missions. Of primary importance was the Synchronous Equatorial Orbiter (SEO) derivative of Lunar Orbiter; while, of secondary importance was the Mars Orbiter (MO). The one-year SEO and the MO baselines were readily derived from the Lunar Orbiter and were used during the parametric analyses. Descriptions of the synthesization of these payloads are contained in this section.

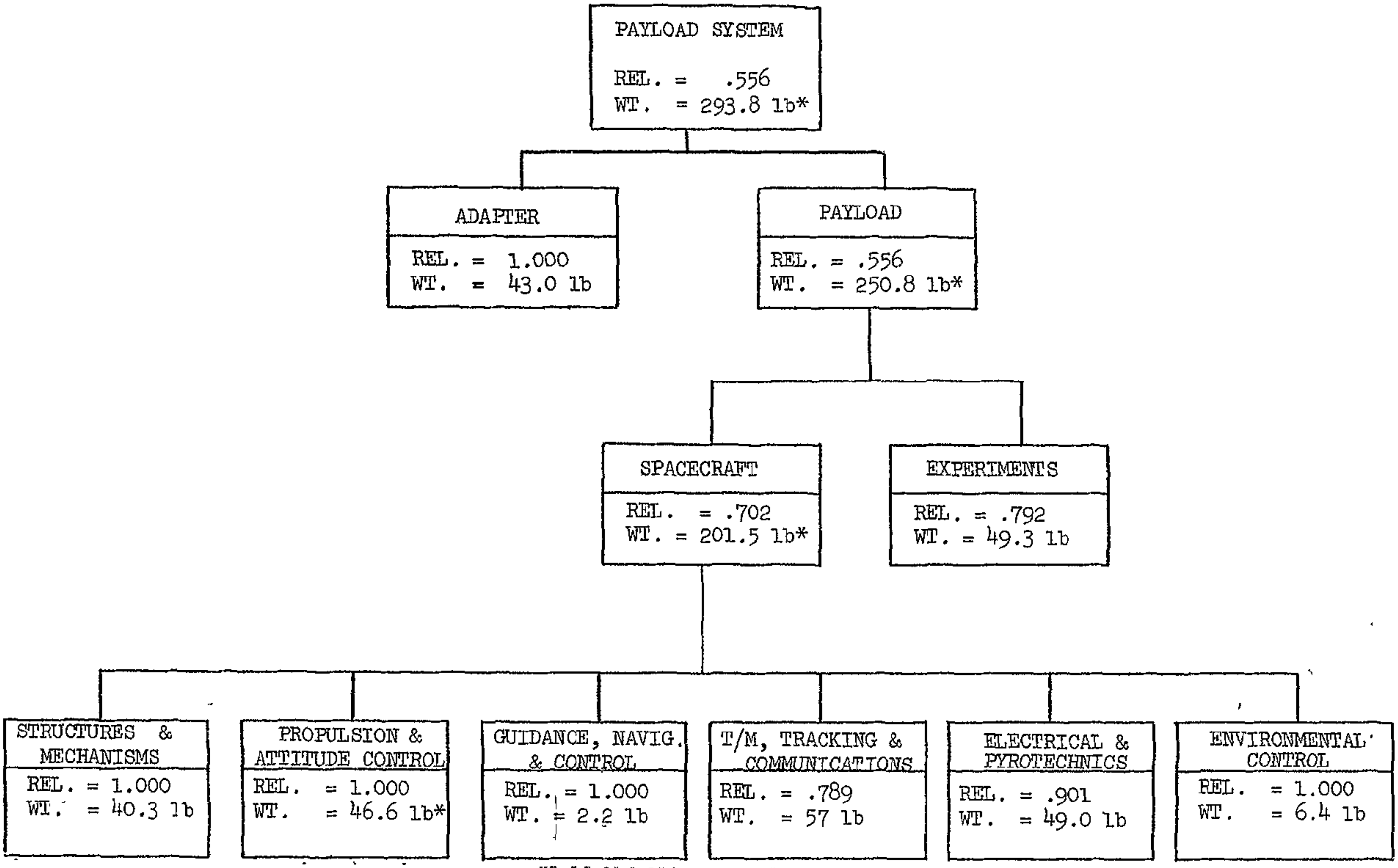
Following completion of the parametric analyses, NASA and Aerospace requested that the one-year SEO be modified to a two-year configuration in order to provide better correlation with the traffic model. Changes to the one-year configuration required for a two-year mission are also discussed.

PAYLOAD: SRS - BASELINE MISSION: EARTH ORBIT

($ IN MILLIONS - 1970 $)

SUBSYSTEM \ COST CATEGORY	NON - RECURRING COSTS					RECURRING COSTS					TOTAL PROGRAM COST
						HARDWARE			OPERATIONS		
	DEVEL.	GSE	S/C INT. & TEST	PROG. MGMT.	TOTAL	QTY.	AVE. UNIT	TOTAL	AVE. UNIT	TOTAL	
PAYLOAD ASSY. AND INTEGRATION	.89	-			.89	4	.10	.40	.02	.08	1.37
EXPERIMENTS AND MISSION PECULIAR EQUIPMENT	.36	-			.36	4	.08	.32	.11	.44	1.12
STRUCTURES AND MECHANISMS	.92	.03			.95	4	.07	.28	.02	.08	1.31
ELECTRICAL AND PYROTECHNICS	2.08	.37			2.45	4	.34	1.36	.02	.08	3.89
GUIDANCE, NAVIGATION, STABIL., AND CONTROL	.09	-			.09	4	.03	.12	-	-	.21
PROPULSION AND ATTITUDE CONTROL	.45	-			.45	4	.09	.36	-	-	.81
TELEMETRY, TRACKING, AND COMMAND	2.60	.45			3.05	4	.43	1.72	.02	.08	4.85
ENVIRONMENTAL CONTROL	.15	-			.15	4	.02	.08	-	-	.23
SUB TOTAL — ALLOCATED	7.54	.85			8.39	4	1.16	4.64	.19	.76	13.79
SUB TOTAL — NON-ALLOCATED TO SUBSYSTEM	.74	-			.74	4	.23	.92	-	-	1.66
PAYLOAD TOTAL	8.28	.85			9.13	4	1.39	5.56	.19	.76	15.45

Fig. 3-15 PROGRAM COST APPORTIONMENT - SRS



* Weights include 29 lb of solid propellant (for orbit positioning motors)
(Note: All reliability numbers based on probability of 6 month operation at 50% confidence level.)

1 lb = 0.4536 kg

Fig. 3-16 RELIABILITY AND WEIGHT APPORTIONMENT
BASELINE SRS - ALTERNATE 1 (VHF-FIXED SPIN AXIS)

The SRS baseline was considered to be a typical inexpensive space physics payload; however, examination of the traffic model indicated that, with minor modifications, the SRS could be redesigned to make it a closer match to planned NASA magnetosphere payloads. Changes to the SRS design for this new mission with revised weight, cost and reliability data are provided in Par. 3.3.4.

Following the parametric analyses, it was agreed by NASA that there was insufficient time to fully investigate the Mars Orbiter, hence MO did not undergo redesign and costing. Thus, the final payload selections for conceptual low-cost redesign and costing were the one-year OAO-B, two-year SEO, and the 6-month SRS.

3.3.1 Synchronous Equatorial Orbiter (One-Year) Baseline Data

3.3.1.1 General Description. The Synchronous Equatorial Orbiter (SEO) is a modified Lunar Orbiter spacecraft that incorporates a number of fundamental changes to the basic spacecraft to make it capable of performing the synchronous earth-resources mission. The synthesized baseline SEO is shown in Fig. 3-17.

3.3.1.2 Mission Description. The objective of the SEO program is to obtain selected medium resolution photographs and continuous low resolution imagery of the earth's surface in the visible spectrum. The baseline spacecraft was designed for a 1-year lifetime. Four operating spacecraft in orbit will assure total earth coverage. As planned, the SEO is launched by an Atlas SLV-3A/Agena/Burner II into synchronous equatorial orbit.

3.3.1.3 Subsystem Descriptions. The following subsystem changes to the baseline Lunar Orbiter were required because of new mission requirements, differing environment and extended life. In addition, certain components not required for the synchronous mission were deleted. The subsystem changes are shown in Fig. 3-18.

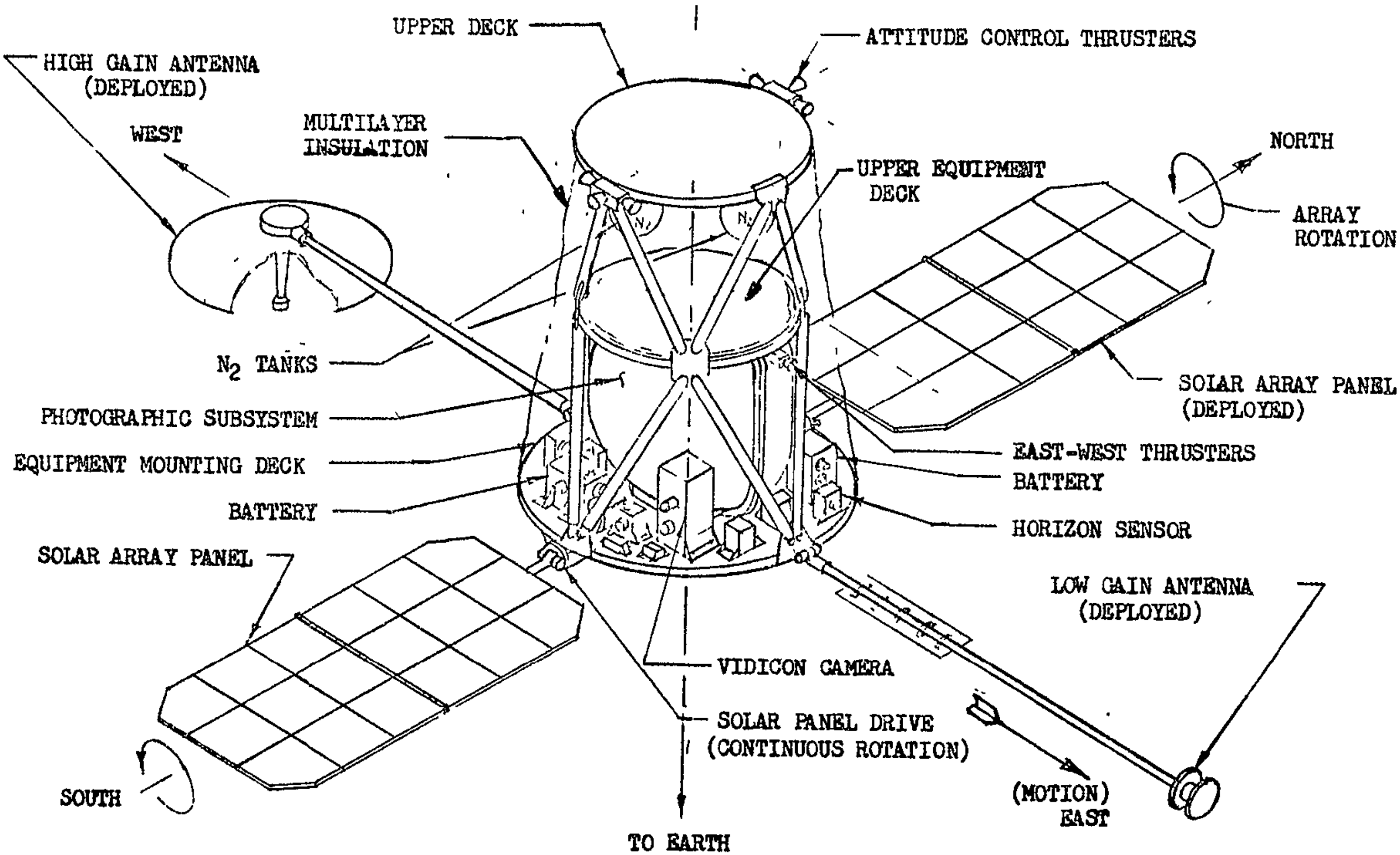


Fig. 3-17 Baseline SEO Configuration

SUBSYSTEM	DELETE	CHANGE	ADD
Experiment	• Wide Angle Camera Lens • Image Motion Compensation	• Increase film qty. and bimat qty.	• TV Camera • Secondary Experiments (from CDPI)
Structures & Mechanisms	• Insulation • Elec. cabling	-	• Array storage devices (from elec.)
Environmental Control	-	-	• Insulation (from Structure)
Propulsion	• Total velocity control system (bi-propellant)	-	-
Attitude Control (N_2 cold gas)	-	-	• Reaction Control Sys. (from G&N)
Electrical Power	-	-	• Cables (from Struct) • Sun Sensor • S/A Orientation Device
Guidance & Navigation	• Canopus Tracker • Reaction Cont. Sys. • IRU	-	• Reaction Wheels • Earth Horiz. Sensor • Polaris Tracker
Communications, Data Processing, Instrumentation	• Secondary Experiments	• Increase TWTA qty.	• D/A Converter • Tape Recorder

Fig. 3-18 Translation of Lunar Orbiter to SEO 1-Yr. Baseline

Adapter - No changes were made as the baseline launch vehicle remains the same.

Experiments - Changes were made to the Photo Payload to compensate for increased lifetime and change in photographic altitude. These changes are:

- Low resolution (80 ℓ/mm) optics and related mirrors were deleted.
- Rewind was deleted and the readout scheme changed.
- The V/H Sensor was deleted.
- The Optical-Mechanical Scanner was made standby redundant.
- Film was changed to EK 3404 and the film web load was quadrupled.
- Drive motors were made parallel redundant.
- The Nimbus AVCS was added for low resolution TV coverage.

The baseline Lunar Orbiter's primary payload was the photographic subsystem consisting of a high resolution and a medium resolution lens system, a single camera housing, on-board film processor and an optical mechanical scanner and readout system. The high resolution camera is capable of resolving 1 meter from an altitude of 46 km at the moon.

At synchronous orbit, the payload is stationary over a given point on the earth's surface. Therefore, there is no relative motion between it and the earth and no requirement for either V/H or Crab Attitude Sensors and Controls. Also, because of the long distance of the sensor to the earth, only a long focal length lens will provide useful data. A 9 in. (0.23 m) focal length lens, was chosen for SEO. The field of view of this lens, across the 0.55 m width of the film, covers one-quarter of the earth's equator from synchronous orbit. A square format was chosen with 45° coverage into both the northern and southern hemispheres. Ground resolution, based upon lens resolution of approximately 80 lines per mm, is 1.24 nm (2.28 km).

In order to obtain maximum photographic coverage, photography will start as soon as a sufficient portion of the earth's surface is illuminated. Approximately eighty percent of the earth's surface within the camera field of view is illuminated at 0800 hours. At 1600 hours, the illuminated portion of the

earth's surface has again been reduced to eighty percent and the last frame for the day is exposed. In addition, frames are exposed at 1000, 1200, and 1400 hours local time for a total of five exposed frames per day. Since the film takes a permanent set if it sits on the rollers for longer than 8 hours, and permanent lamination of the Bimat and film occurs if they remain in contact longer than 15 hours, the film will be moved one complete frame at 2400 hours. The film and Bimat consumption will be six frames per day for a total of 452 ft (138 m) for a one-year mission. The film and Bimat spool diameters were increased.

Bimat storage is a problem because of its limited storage life. The most suitable storage conditions are a high moisture atmosphere with temperature maintained in the 0° to 4.4°C range. The Bimat supply system was redesigned to maintain these conditions. In order to facilitate the temperature control of the Bimat supply, the Photographic subsystem must be thermally isolated from the remainder of the spacecraft and stabilized in the temperature range of -6.7° to 0°C. Thermostatically controlled heaters are used to maintain the temperature of the Photographic payload.

The rest of the Photographic subsystem is made up of simplified modifications of the Lunar Orbiter Photographic Subsystem components. A between-the-lens shutter has been substituted for the focal plane shutter to protect the film and platen from damage when the sun is seen by the camera.

Another consideration influencing experiment redesign is potential radiation fogging of the film. For a one-year mission at the time of a solar maximum (next period 1979-1980), the integrated proton dose would be about 100 rads. This would result in a density increase in EK-3404 film of 0.48 with a shielding thickness of 0.85" (27.6 m/m) of aluminum. This is an acceptable density increase with minimal loss of resolution. The probability of this occurring is 1 in 10 in the high event solar years. In the quiet sun years, the integrated dose would be an order of magnitude less at the same probability. Trapped electronc and protons are not a problem at this altitude as the environment is essentially the interplanetary medium. Radiation is not a serious constraint for this mission.

At the suggestion of NASA Headquarters, a supplemental TV type imaging device is included to increase the mission utility. The selected system was the Nimbus Advanced Vidicon Camera System (AVCS). This system could be used for broad weather coverage and for identifying phenomena for subsequent high resolution photography.

Structures and Mechanisms - Structural weight was increased to accommodate additional experiments, propellant and equipment. A slight increase in thickness of the shielding of the film storage was included.

Environmental Control Subsystem - No significant changes were made as the environment is essentially the same as that at the moon. Some insulation was added.

Propulsion and Attitude Control - Redundant cold-gas system components were added to increase lifetime. The gas supply was increased by adding a second N_2 tank. East-West thrusters were added for orbital maneuvering. The LO propulsion subsystem was removed as it is not required.

Electrical Power - The four solar arrays were replaced with two large solar arrays. The same solar array area was maintained. The solar arrays were redesigned to permit sun tracking. Redundancy for certain items was provided.

Guidance and Navigation - For the Earth operation this subsystem was modified by the replacement of the Canopus tracker by a horizon sensor and a Polaris tracker. Orbital stability is provided through the use of reaction wheels. The inertial reference unit (IRU) was removed as not required.

CDP&I - Antenna position control was deleted and a wide band tape recorder was added to record AVCS data. Associated electronics were also included.

3.3.1.4 Synchronous Equatorial Orbiter Cost Estimate. The cost estimate for the synthesized SEO was derived from the basic Lunar Orbiter cost data as allocated by LMSC from the Boeing NASA 533 forms. In general, adjustments were made to subsystem costs by adding or eliminating component costs, where such were identifiable, or estimating, using the average \$/lb derived from Lunar Orbiter data.

Specifically by subsystem the following major adjustments were made:

Experiment Subsystem	- Remove V/H Sensor at \$3.826M - Remove 80 mm Lens at est. \$660K based on average \$/lb photo S/S - Add Nimbus AVCS at \$66K unit cost and \$12M R&D cost - Add Optical-Mechanical scanner at \$1.527M - Changed the 24" (0.61 m) focal length lens to a 9" (0.23 m) focal length lens
Structures Subsystem	- Costed at the LO average \$/lb resulting in a 10 percent increase of R&D and unit cost.
Propulsion and Attitude Control	- Unit cost was adjusted to reflect deletion of propulsion. Attitude control was increased for the 1-year mission.
Power Subsystem	- Cost additions include \$120K/unit for the sun tracking mechanisms and \$273K/unit for the additional power. The development cost reflects additional \$250K for the sun trackers and \$390K for the additional power integration.
G&N Subsystem	- The changes include addition of a \$165K Polaris Star Tracker, \$15K earth horizon sensor, \$200K worth of reaction wheels and removal of Canopus Star Tracker at estimated \$346K. The development cost and test cost were also reduced to reflect the R&D efforts involved in modification of the Canopus Star Tracker.

CDPI Subsystem	-	Estimated 10 percent increase in the RDT&E costs. The unit cost increased by $66K for the additional tape recorder and D-A converter and $41K for the extra TWTA and PCM Multiplexer/Encoder.
Adapter	-	Since most of the components added were off-the-shelf items and no additional development cost was charged to the subsystems, the payload assembly and integration development cost was increased on basis of extrapolating the LO costs to the higher weight SEO. The extrapolation was performed across the board to allow for the unit integration and additional operations costs. The unit cost was then reduced by removing the cost of LO shroud estimated at $263K. The shroud development of $3.048M is also excluded from the non-recurring costs and the spacecraft assembly costs of $2.876M were shifted to structures S/S development.

The non-recurring costs derived above from the individual adjustments at the subsystem level were checked against the in-house CERs as shown in Fig. 3-19.

The in-house CERs for operations do not reflect the costs of one year photographic mission, thus the $8M estimate is based solely on Lunar Orbiter data extrapolations. These consisted of quadrupling operating costs to reflect four (4) times the mission duration and minor delta costs in ACS, CDPI and SCS to represent a larger satellite size. The SEO baseline costs are shown in Fig. 3-20 for a four flight-unit program. It should be noted that these costs include the propulsion subsystem and Inertial Reference Unit which were subsequently removed by agreement with Aerospace. As the one-year configuration (see Section 3.3.3) was not used, the baseline costs reflecting these changes were not revised.

3.3.1.5 <u>Synchronous Equatorial Orbiter (SEO) Weight and Reliability Estimate</u>. The summary of subsystem, payload, and payload system weights and reliabilities for the baseline SEO are tabulated on Fig. 3-21. The 735 lbs (334 kg) (including attitude control gases) is the total weight of the payload as it separates from the upper stage Agena. The reliability of 0.730 represents the probability of satisfactory operation in orbit for one year without catastrophic failure.

Development Cost	In-House CER Estimate	Sync. Eq. Orbiter Estimate
Structures & S/C Assembly	$ 4.85 Million	$ 3.74 Million
Environment Control		0.06
Adapter & Integration	1.61	1.61
Subtotal	$ 6.46 Million	$ 5.41 Million
Power	$ 4.50	$ 4.95
G&N	4.50	3.35
Propulsion/Att. Control	2.60	1.43
TT&C	10.90	9.01
Photo (no in-house CER)	30.26	30.26
Subtotal	$ 59.22	$ 54.41
Unallocated	0	5.00
TOTAL S/S DEVELOPMENT	$ 59.22	$ 59.41
GSE - Photo (no in-house CER)	$ 7.75	$ 7.75
- S/C	18.85	19.01
S/C Assembly & Integration	19.50	20.83
Program Management	8.43	7.34
TOTAL NON-RECURRING	$113.75	$114.34

Unit Cost	In-House CER Estimate	Sync. Eq. Orbiter Estimate
Structure	$ 372.9 Thousand	$ 346.8 Thousand
Environmental Control		16.5
Adapter	1,300.0	217.1
Spacecraft Assy. & Integ.		1,264.5
Subtotal	$ 1,672.9	$1,844.9
Electrical (built up)	1,655.6	1,655.6
TT&C	2,290.0	2,258.3
G&N	2,487.0	2,442.2
Propulsion/Att. Control	651.0	425.1
Experiments (no CER)	2,074.4	2,074.4
Subtotal	$10,830.9	$10,700.5
Unallocated	0	240.0
TOTAL UNIT	$10,830.9	$10,940.5

Fig. 3-19 1-Year SEO Cost Comparison

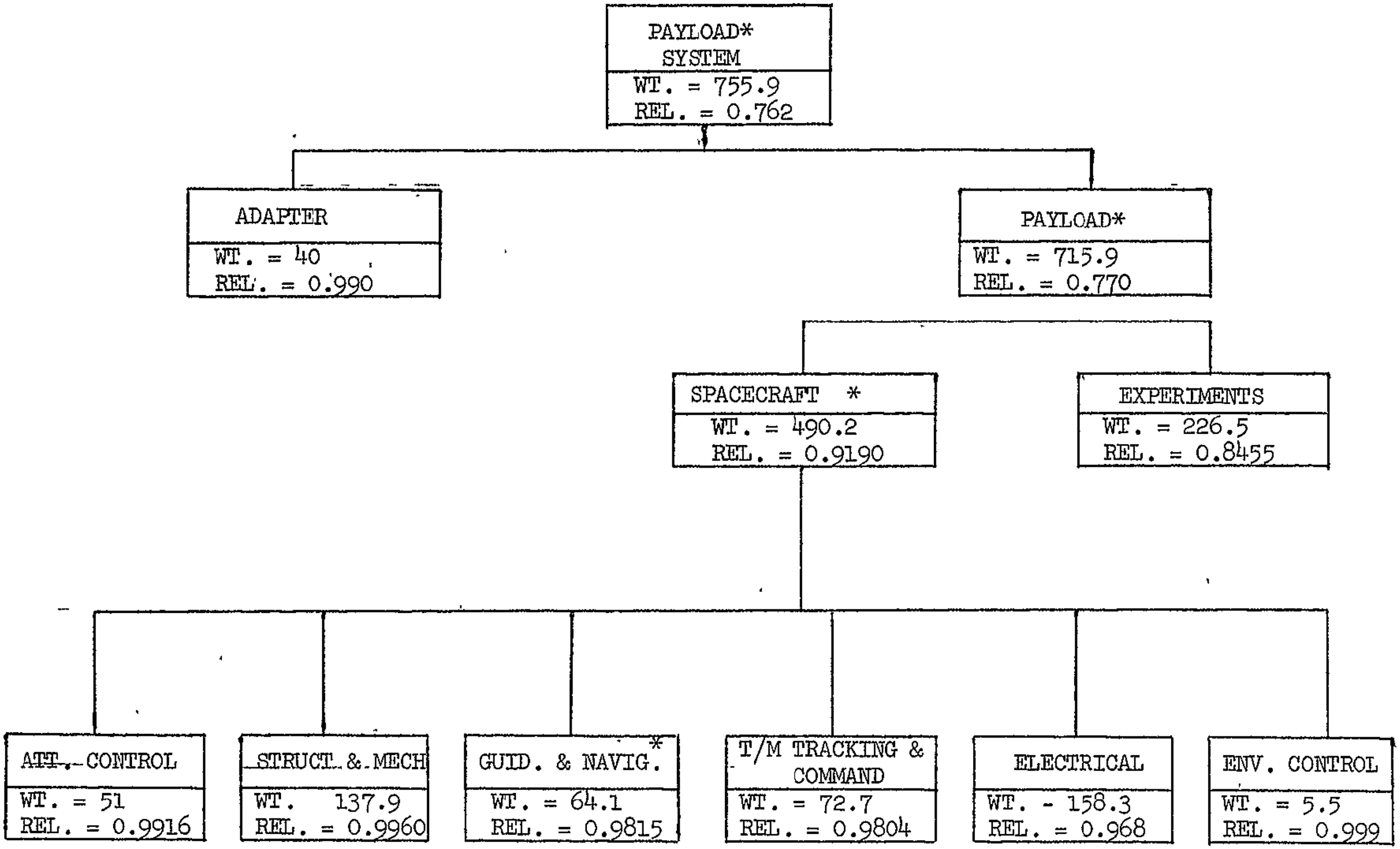
PAYLOAD: SYNC. EQ. ERS - BASELINE MISSION: SYNCH. EQ. ORBIT (1 YEAR)

($ IN THOUSANDS) - 1970 $

SUBSYSTEM \ COST CATEGORY	NON-RECURRING COST					RECURRING COSTS					TOTAL PROGRAM COST
						HARDWARE			OPERATIONS		
	DEVEL.	GSE	S/C INT. & TEST	PROG. MGMT.	TOTAL	QTY	AVE. UNIT	TOTAL	AVE. UNIT	TOTAL	
ADAPTER	$1613.4 (1)	-	$1101.2	$ 172.8	$ 2887.4	4	$ 217.1 (2)	$ 868.4	$640.9	$2563.6	$6,319.4
EXPERIMENTS AND MISSION PECULIAR EQUIPMENT	30263.3	7751.2	5948.6	2608.0	46571.1	4	2074.4	8297.6	3507.4	14029.6	68,898.3
STRUCTURES AND MECHANISMS	3740.7	1013.9	2340.2	229.2	7324.0	4	1611.3	6445.2	327.7	1310.8	15,080.0
ELECTRICAL AND PYROTECHNICS	4954.2	1708.4	2039.9	1653.5	10356.0	4	1655.6	6622.4	613.0	2452.0	19,430.4
GUIDANCE, NAVIGATION, STABIL., AND CONTROL (4)	3345.9	4409.2	4170.4	644.0	12569.5	4	2442.2	9768.8	872.7	3490.8	25,829.1
PROPULSION (4)	762.1	1634.0	1525.0	211.7	4132.8	4	252.0	1008.0	307.5	1230.0	6,370.8
ATTITUDE CONTROL	672.6	490.1	610.2	95.8	1868.7	4	173.1	692.4	63.8	255.2	2,816.3
TELEMETRY, TRACKING, AND COMMAND	9010.9	9752.8	3044.7	1722.9	23531.3	4	2258.3	9033.2	985.0	3940.0	36,504.5
ENVIRONMENTAL CONTROL	55.0	-	45.0	5.0	105.0	4	16.5	66.0	5.0	20.0	191.0
SUB TOTAL — ALLOCATED	54418.1	26759.6	20825.2	7342.9	109345.8	4	10700.5	42802.0	7323.0	29292.0	181,439.8
SUB TOTAL — NON-ALLOCATED TO SUBSYSTEM	5000.0	-	-	-	5000.0	4	240.0	960.0	670.1	2680.4	8,640.4
PAYLOAD TOTAL (3) (4)	59418.1	26759.6	20825.2	7342.9	114345.8	4	10940.5	43762.0	7993.1	31972.4	190,080.2

Notes: (1) Excludes $3.048M for Shroud Development/Mods.
(2) Excludes Cost of Shroud Est. at $263,400.
(3) Excludes Prime Contractor Fee & NASA Program Mgmt.
(4) Includes propulsion and IRU which were subsequently removed (see Section 3.3.3)

Fig. 3-20 Program Cost Apportionment - Sync. Eq. Earth Resources Satellite (Lunar Orbiter Derivative)



* Reflects removal of propulsion subsystem and IRU.

1 lb = 0.4536 kg

Fig. 3-21 Baseline Sync. Eq. Orbiter (Earth Resources) - 1 Yr. Reliability & Weight Apportionment

The variation in payload weight and reliability to that shown in Section 3.1 reflects removal of the IRU and propulsion subsystem by mutual agreement with Aerospace. These weights and reliabilities are those that were used in the parametric analysis and SEO optimization, but not in development of subsequent low-cost designs as this was based upon the 2-year configuration.

3.3.2 Mars Orbiter

3.3.2.1 General Description. The second Lunar Orbiter derivative was the Mars Orbiter (MO). As with SEO, mission parameters and location are changed necessitating design changes. Principal changes involve the power system due to reduced solar illumination at Mars, the propulsion system due to increase in velocity required for attaining Mars orbit, and CDPI due to greatly increased communications distances. The MO is shown in Fig. 3-22.

3.3.2.2 Mission Description. The selected mission for this configuration is the 1971 Mars orbit. The overall experiment objectives are to provide a photographic atlas of the planet Mars and to provide high resolution photography of candidate Mars manned landing sites. Supplemental experiments to increase mission utility were derived from the Mariner Mars Orbiter mission which will provide UV and IR imaging at Mars, and micrometeoroid and radiation measurements enroute and at Mars.

The spacecraft is launched by an SLV-3C/Centaur. 1971 is an ideal year for Mars flights as the planet is at perihelion, thus the velocity required is less than nominal. In the reconfiguration, this was considered and resulted in the injected weight to the mission velocity being less than the launch vehicle's capability. For the 1971 mission, the required injection velocity is 37,400 fps (11,400 m/sec). Hyperbolic speed at Mars is 9705 fps (2958 m/sec). The launch window is between 15 May and 2 June 1971 with arrival window at Mars being 3 December 1971 to 6 January 1972. Communication distance will vary from 0.96 A.U. at arrival to 2.0 A.U. at end of readout. The maximum tine enroute is 218 days, the end of the window. The launch vehicle is capable of boosting 2175 lbs (989 kg) to this mission.

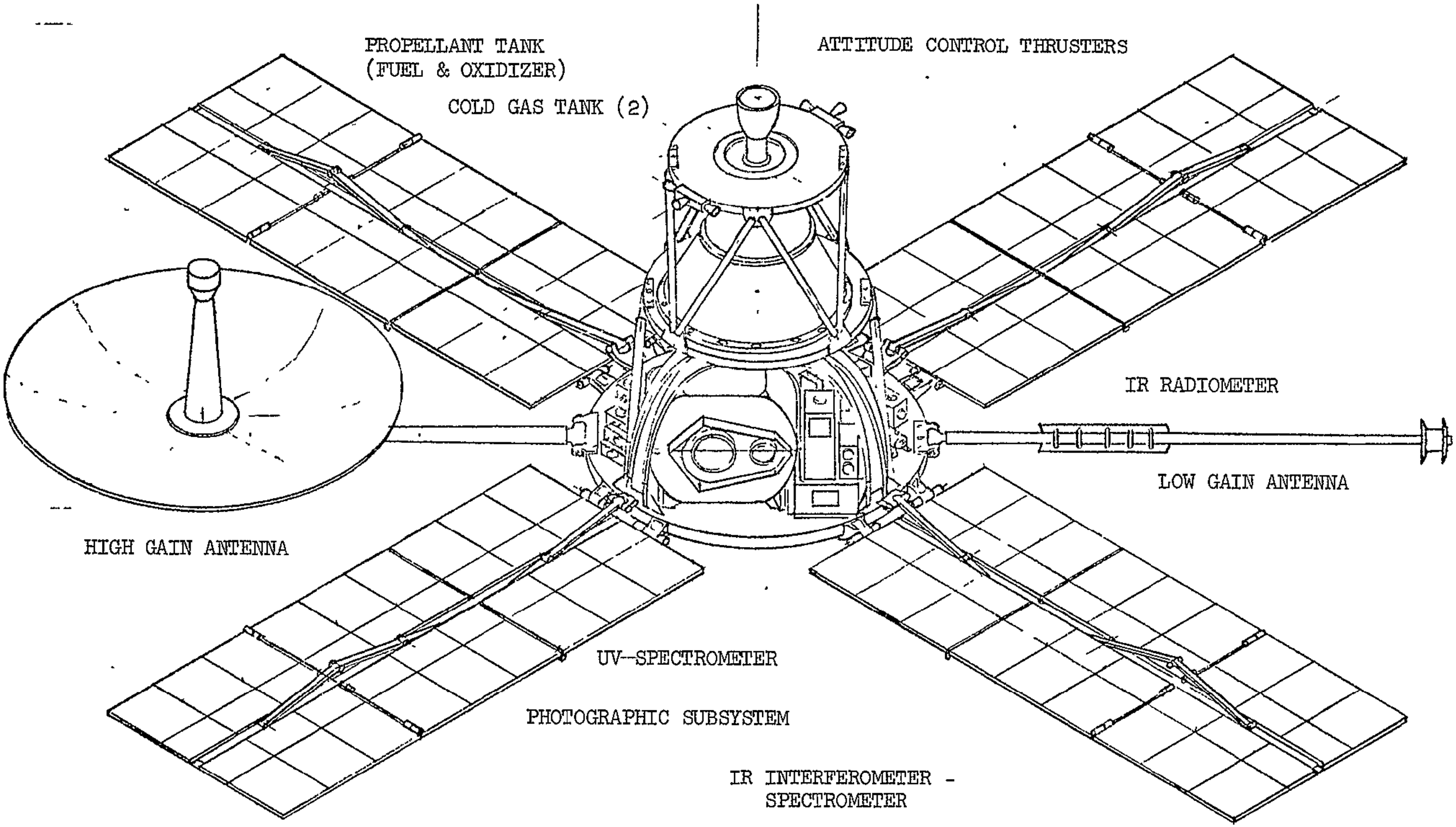


Fig. 3-22 General Arrangement of Baseline Mars Orbiter

At arrival at Mars, the spacecraft, having been targeted for a pericenter of 1.5 radii, will be braked to an elliptical orbit of 1.5 by 5 radii. This will require 4789 fps (1457 m/sec). After orbit verification, the pericenter will be lowered to 1.06 radii by further retrofiring at apocenter - 458 fps (140 m/sec). The midcourse maneuvering is estimated at 400 fps (122 m/sec) resulting in a total ΔV of 5647 fps (1720 m/sec). Using the existing Lunar Orbiter propulsion system with I_{sp} of 273 secs results in a total weight to inert weight ratio of 1.90093. Orbit lifetime at the selected orbit is about 120 days. A sufficient quantity of residual velocity is available to raise pericenter if required. The orbital period is 8 hrs and 7 min, and the eccentricity is 0.65. Three pericenter passes per Mars day can be used for high resolution photography. Inclination should be selected to maximize the coverage.

Operations in Mars orbit are planned to be as similar in concept as possible to the Lunar Orbiter mission. However, changes in operation time due to increased coverage requirements and communications readout are required.

3.3.2.3 Subsystem Description

Experiments - The Lunar Orbiter's photographic payload is considered to be satisfactory for the Mars mission with the exception of the film loading and sequence of processing. To circumvent bonding of web and film, additional film and web (and storage capacity) must be provided to permit processor creep after initiation of photography. In view of longer readin/readout cycles and increased mission duration, a redundant optical/mechanical scanner was added and supplemental gas was provided. Secondary IR and U/V experiments from Mariner '71 were added to increase mission utility.

The baseline payload for the Lunar Orbiter was optimized for a 30-day mission with 20 days of photography and ten days readout. Mission duration and communications distance force some modifications to the payload for the Mars mission. Link limitations, coupled with increased pericenter distance, cause some reduction in ground resolution distance (GRD). The high resolution system on LO was capable of 1 meter resolution. Raising the altitude at pericenter to 110 nm

(202 km) reduces GRD by a factor of 4. The CDPI problem revolves around loss in signal-to-noise ratio at the interplanetary distance. Improving CDPI performance by increasing transmitter power and antenna aperture and by using the 210 ft (64 m) DSIF antennas reduces path losses from 53 db to 32.5 db (1.736 times reduction). Assuming all other parameters remain the same, it would result in a transmission time of 55 days per frame or 29 years to readout a complete 194 frame payload. This is impractical. The only reasonable alternative would be to accept reduced resolution, and, by restricting the transmission time to about 12 hours per frame, a GRD of 30 meters is possible and reasonably practical. This is 3 times better than that predicted for Mariner '71. Extended readout times, 90 days to readout the 194 frames, necessitates adding a redundant Optical-Mechanical Scanner. Additional film is provided to allow for film advance during quiet periods after arrival to prevent film/web bonding. No problem is envisioned enroute; but after arrival, film must be moved through the processor each eight hours or film breaking could result. The same mission procedure as the Lunar mission is planned with respect to readin and readout except that extended frame transmission times will require a much longer readout period. For the Mars mission, readin is planned for the first 30 days and then readout for the following 90 days, as opposed to 20 days readin and 10 days readout at the moon. Film fogging due to radiation is less of a problem than in synchronous equatorial and no special shielding provisions are planned. The N_2 supply in the Photo Subsystem is doubled to compensate for leakage and film handling size increased to compensate for the increased film/web supply.

In the area of secondary experiments, additional sensors similar to Mariner '71 were added to increase mission utility. These included an Infrared Interferometer/Spectrometer, an Infrared Radiometer, and an Ultra-Violet Spectrometer. A wide band tape recorder is provided to store secondary experiment data.

Adapter - Use of the SLV-3C/Centaur necessitates a different payload adapter.

Structures and Mechanisms - Size and weight capability of the tank and equipment mounting decks and strength of the truss structure were increased. The high gain antenna and solar array deployment mechanisms required modification.

Environmental Control - The deep space environment necessitated an increase in the capability of the ECS. Heater capacity and insulation were increased.

Propulsion - The main change was to the propellant storage caused by increased loading. A single common bulkhead tank was substituted for the existing tanks; a differential tank pressure regulator was added.

Attitude Control - The longer mission requires more control gas. A second N_2 tank and associated plumbing were added.

Electrical Power - At Mars, solar intensity is down by a factor of 2.8. Additional power for transmitters and heaters is required. Thus, the array area was increased by a factor of 3.3. Redundant components and a second battery were added.

Guidance and Navigation - Reliability of the existing inertial reference unit is not satisfactory for the mission, hence a more reliable unit was substituted.

CDPI - A 9 ft (0.27 m) aperture extendable high gain antenna was substituted for the existing antenna, and redundant 40 watt TWTAs were provided to increase system gains. Redundant transponders and PCM multiplexers and a tape recorder were added.

3.3.2.4 Mars Orbiter Cost Estimate. The Mars Orbiter costs were extrapolated from the basic Lunar Orbiter data in the similar manner to the Synchronous Equatorial Orbiter.

The major cost adjustments at the subsystem level are as follows:

Experiment Subsystem - Add Optical Mechanical Scanner, film, and web capacity at a total cost of $3M (64-67 $). Also add secondary experiments resulting in $6M additional development costs including $1.6M of GSE and $2.4M of spacecraft integration and test. The unit cost of secondary experiments was

estimated at \$2.2M. The major cost increase for this subsystem resulted in operations due to the mission requirements, such as mission duration of 338 days as compared to Lunar Orbiter's 34 days. The readout duration of 90 days as compared to 10 days for Lunar Orbiter. Thus, the photo system operating costs of Lunar Orbiter were factored 10 times to derive the Mars Orbiter system operating costs. Also, the secondary experiments resulted in additional operating costs estimated at \$2.9 million per unit.

Structures Subsystem - The structures cost was increased based on CERs and increased weight.

Propulsion Subsystem - This subsystem is the same as in the SEO case and the costs reflect use of a single aluminum tank, slightly larger than in the SEO. The costs were estimated using SEO \$/lb with no change for the GSE and integration portions. Again, the attitude control subsystem is broken out separately with the same RDT&E cost as for SEO, but a slightly reduced unit cost based on \$/lb from SEO.

Power Subsystem - The costs were estimated using a CER \$/watt for the enlarged solar arrays plus the additional battery and cabling costs-estimated on \$/lb basis. Development cost was also increased based on \$/lb CER data.

Guidance and Navigation Subsystem - The development cost was assumed unchanged, the unit cost was derived using LO and closely correlating CER \$/lb to reflect slightly heavier system than the LO subsystem.

CDPI Subsystem - Since most of the additional weight was due to redundancy with the exception of a new antenna and a recorder not in the LO CDPI subsystem, the development and integration costs were increased 15 percent keeping the GSE cost the same. The unit costs were derived using CDPI CER and cross-checked by summing the LO unit cost and the redundant/additional components. Operating costs were increased based on the ratio of unit costs.

Adapter Subsystem - This includes only the adapter whose new-development was estimated at $20K/lb ($9K/kg). The unit cost was estimated at $300/lb ($1364/kg) and the support and management costs were allocated resulting in $348K/unit.

The non-recurring costs estimated for the Mars Orbiter based on a combination of Lunar-Orbiter, Synchronous Equatorial Orbiter and in-house CERs were checked against in-house CER estimates with the resulting variances as shown in Fig. 3-23.

The operating costs were extrapolated directly from the Lunar Orbiter and Synchronous Equatorial Orbiter data with a major adjustment for the experiment subsystem as discussed above.

The costs in Fig. 3-24 are the Mars Orbiter estimates derived as explained above and summarized into a five flight-unit program.

3.3.2.5 Mars Orbiter Weight and Reliability Estimates. The baseline Mars Orbiter is also an extrapolation of the reference Lunar Orbiter. The summary reliability characteristics and weights are tabulated on Fig. 3-25. The total payload as injected into trans-Mars trajectory weighs 1,861 lb (846 kg)(including 920 lb (418 kg) of propellants, expendable gases, and residuals). The payload reliability of 0.8028 represents the probability that the Mars Orbiter will complete its mission without catastrophic failure; the mission operating time will be an average 218 days in transit, plus 120 days in orbit about Mars, a total of about 11 months. As in the case of the SEO, the Lunar Orbiter reliability and weight figures were extrapolated to the 11-month Mars mission.

3.3.3 Synthesis of the Long-Life SEO (2-year SEO)

The initial baseline SEO was an extrapolation of the Lunar Orbiter to perform a 1-year mission. To provide increased application to the NASA traffic model, it was later decided (with NASA approval) to establish a baseline SEO with

Subsystem Development Cost	In-House CER Estimate	LO/SEO/CER Based Estimate
Structures and S/C Assy.)	$ 5.147 Million	$ 4.250 Million
Environmental Control)		0.060 "
Adapter and Integration	2.020 "	2.020 "
Subtotal	$ 7.167 "	$ 6.330 "
Power	5.000 "	5.242 "
G&N	4.000 "	3.606 "
Propulsion)	2.600 "	0.875 "
Attitude Control)		0.673 "
TT&C	12.470 "	9.420 "
Photo (no in-house CER)	27.194 "	27.194 "
Subtotal	$ 58.431 "	$ 53.340 "
Unallocated	0	5.000 "
Total S/S Development	$ 58.431 "	$ 58.340 "
GSE - Photo (No in-house CER)	9.091 "	9.091 "
- S/C	20.800 "	19.073 "
S/C Assy. & Integr.	23.900 "	21.862 "
Prog. Mgmt.	8.978 "	7.354 "
Total Non-Recurring Cost	$121.200 Million	$ 115.720 Million

Unit Cost	In-House CER Estimate		LO/SEO/CER Based Estimate	
Structure)	$ 388.3	Thousand	$ 366.8	Thousand
Environmental Control)			21.5	"
Adapter)	1,495.0	"	347.9	"
Spacecraft Assy. & Integ.)			1,012.1	"
Subtotal	$ 1,883.3	"	$ 1,748.3	
Electrical	$ 1,703.5	"	$ 1,831.2	"
TT&C (used CER for both)	3,307.2	"	3,307.2	"
G&N	2,074.0	"	2,440.0	"
Propulsion	625.0	"	293.8	"
Att. Control	618.8	"	167.3	"
Experiments (no CER)	4,669.8	"	4,669.8	"
Subtotal	$ 14,881.6	"	$ 14,457.6	"
Unallocated	0		200.0	
Total Unit Cost	$ 14,881.6	"	$ 14,657.6	"

Fig. 3-23 Mars Orbiter Cost Comparison

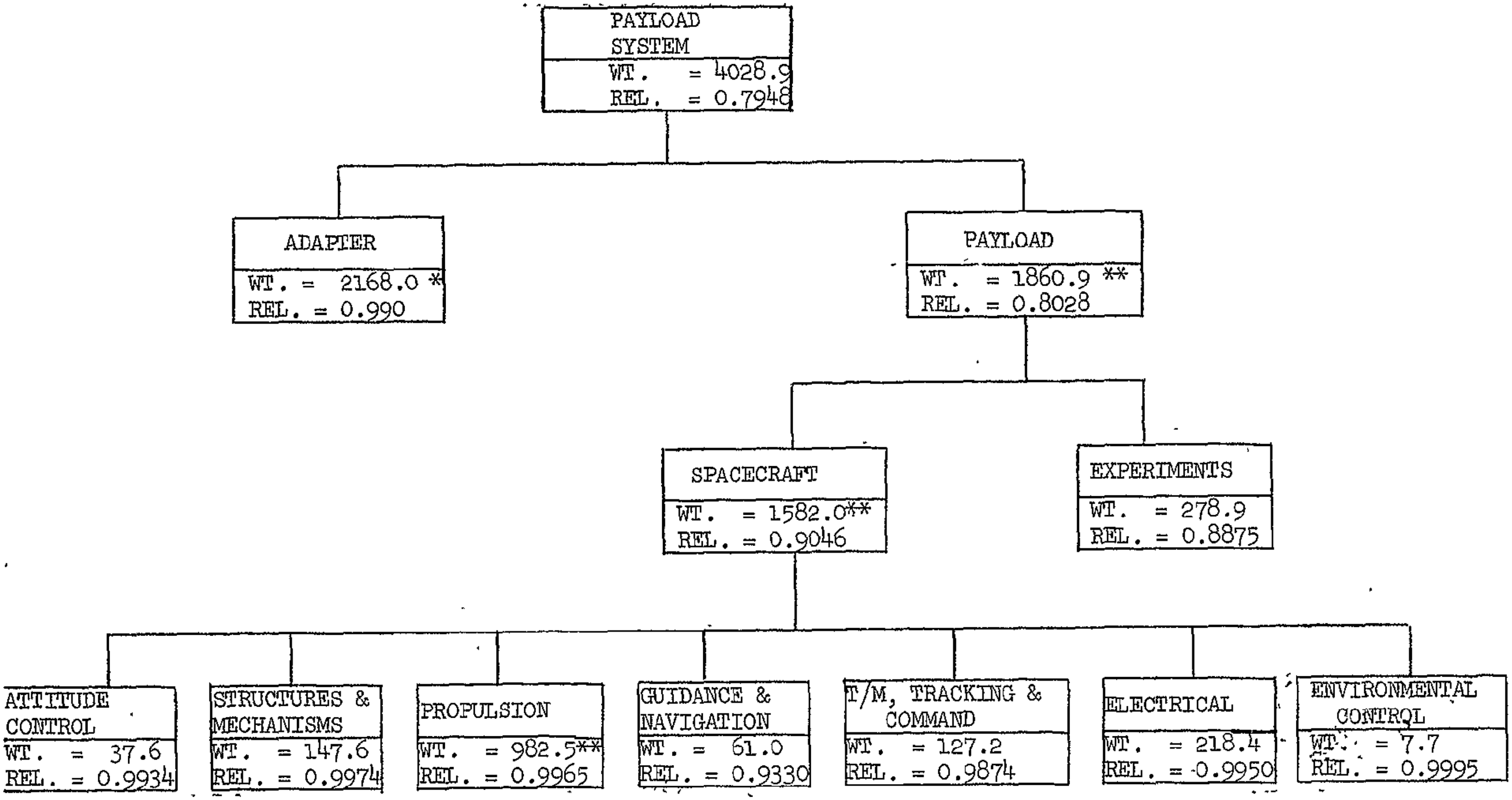
PAYLOAD: MARS ORBITER MISSION: MARS ORBIT - 1 YR.

($ IN THOUSANDS) - 1970 $

SUBSYSTEM \ COST CATEGORY	NON-RECURRING COST					RECURRING COSTS					TOTAL PROGRAM COST
						HARDWARE			OPERATIONS		
	DEVEL.	GSE	S/C INT. & TEST	PROG. MGMT.	TOTAL	QTY	AVE. UNIT	TOTAL	AVE. UNIT	TOTAL	
ADAPTER	$2020.0	-	$ 817.9	$124.8	$ 2962.7	5	$ 347.9	$1739.4	$589.2	$2945.9	$ 7,648.0
EXPERIMENTS AND MISSION PECULIAR EQUIPMENT	27194.1	9091.2	6788.1	2518.3	45591.7	5	4669.8	23349.0	12612.2	63061.0	132,001.7
STRUCTURES AND MECHANISMS	4250.2	1078.4	2557.9	242.1	8128.6	5	1378.9	6894.6	310.8	1554.1	16,577.3
ELECTRICAL AND PYROTECHNICS	5241.6	1708.4	2039.9	1717.1	10707.0	5	1831.2	9156.0	735.6	3678.0	23,541.0
GUIDANCE, NAVIGATION, STABIL., AND CONTROL	3605.9	4109.2	4290.4	665.7	12971.2	5	2440.0	12200.0	875.0	4375.0	29,546.2
PROPULSION	875.0	634.0	1525.0	217.8	4251.8	5	293.8	1469.0	322.9	1614.5	7,335.3
ATTITUDE CONTROL	672.6	490.1	610.2	95.8	1868.7	5	167.3	836.5	57.5	287.5	2,992.7
TELEMETRY, TRACKING, AND COMMAND	9420.6	9752.8	3183.1	1766.2	24122.7	5	3307.2	16536.0	1447.7	7238.5	47,897.2
ENVIRONMENTAL CONTROL	60.0	-	50.0	6.0	116.0	5	21.5	107.5	5.0	25.0	248.5
SUB TOTAL — ALLOCATED	53340.0	28164.1	21862.5	7353.8	110720.4	5	14457.6	72288.0	16955.9	84779.5	267,787.9
SUB TOTAL — NON-ALLOCATED TO SUBSYSTEM	5000.0	-	-	-	5000.0	5	200.0	1000.0	700.0	3500.0	9,500.0
PAYLOAD TOTAL (1)	58340.0	28164.1	21862.5	7353.8	115720.4	5	14657.6	73288.0	17655.9	88279.5	277,287.9

Notes: (1) Excludes Prime Contractor's Fee & NASA Program Management

Fig. 3-24 Program Cost Apportionment - Mars Orbiter



* Includes Fairing, and Adapter which remains with the injection stage.

** Includes Propellants, Residuals and Gases equal to 920 lbs.

1 lb = 0.4536 kg

Fig. 3-25 BASELINE MARS ORBITER - RELIABILITY AND WEIGHT APPORTIONMENT

2-year life. Certain changes in subsystem and payload reliability were made to obtain the equivalent of the 1-year SEO. The duty cycles for all components were made compatible with the 2-year sync.-eq. orbit mission and redundancies and hardware quantities adjusted. The total inert weight of this payload is 1090 lb (495 kg). The reliability of the baseline SEO is 0.607. Weight and reliability summaries by subsystem are shown in Fig. 3-26.

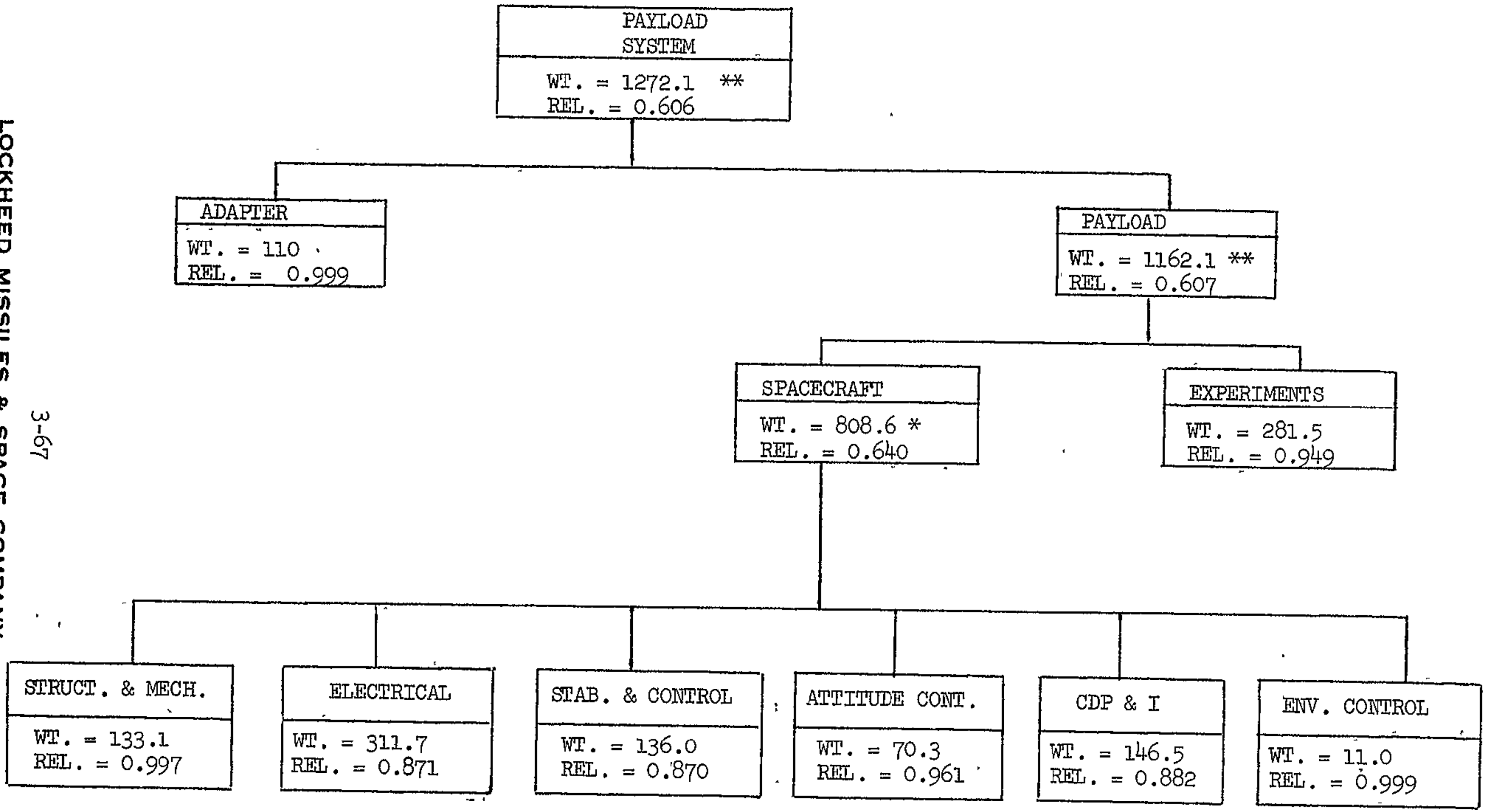
Changes in weight are due to addition of film and bimat plus increased redundancy within the photographic, secondary experiment, environmental control, electrical, CDPI and S&C subsystems. Increased expendable gas for photo payload pressurization and attitude control was provided.

The two-year SEO costs were directly factored from the one-year SEO costs. Addition of redundancy resulted in higher costs. These are shown in Fig. 3-27. It should be noted that the elimination of the propulsion subsystem and the Inertial Reference Unit resulted in some reduction in overall costs. The two-year SEO baseline was subsequently used for the low-cost redesign, costing and planning tasks.

3.3.4 Modification of SRS

The initial baseline SRS was a direct copy of the LMSC P-11 subsatellite. The P-11 spacecraft is designed to be mounted on the aft equipment rack of an Agena vehicle. After the Agena attains orbit, the P-11 separates and is injected into its mission orbit. The P-11 spacecraft includes structure, a solar power system, a command system, a data system and a propulsion system. Space, weight, power, and data handling capability are provided for various types of payloads for a limited operating duty cycle.

The P-11 does not provide sufficient electrical power to perform the HIGLO mission which requires an extended duty cycle. Also, the allocation of equipment to subsystems differs slightly from that which has been established as standard for the Payload Effects Study.



* Does not include 72.0 lbs of expendable gases
** Includes all expendables

1 lb = 0.4536 kg

Fig. 3-26 Baseline 2-Year SEO - Weight and Reliability Estimates

PAYLOAD: SEO BASELINE MISSION: 2 YRS. SYNCH. EQ. ORBIT

($ IN THOUSANDS)

SUBSYSTEM \ COST CATEGORY	NON-RECURRING COSTS					RECURRING COSTS					TOTAL PROGRAM COST
						HARDWARE			OPERATIONS		
	DEVEL.	GSE	S/C INT & TEST	PROG. MGMT.	TOTAL	QTY	AVE. UNIT	5 UNIT TOTAL	2 YEAR MISSION	TOTAL (2)	
Adapter	1613.4		1101.2	172.8	2887.4	5	217.1	1085.5	640.9	2093.7	6066.6
Experiments and Mission Peculiar Equipment	30263.3	7751.2	6498.6	2640.6	47153.7	5	2533.5	12667.5	6313.3	11626.1	71447.3
Structures and Mechanisms	3675.7	1013.9	2340.2	227.1	7256.9	5	1598.3	7991.5	327.7	980.5	16228.9
Electrical and Pyrotechnics	5922.3	2255.0	2369.8	1845.7	12392.8	5	1932.5	9662.5	717.2	2330.4	24385.7
Stabilization & Control	5070.1	4409.2	5577.0	813.0	15869.3	5	3264.2	16321.0	1186.9	3193.7	35384.0
Attitude Control	899.8	588.1	915.3	129.8	2533.0	5	242.5	1212.5	114.8	309.2	4054.7
Command, Data Processing & Instrumen.	10255.0	10252.8	3486.7	1895.5	25890.0	5	3895.0	19475.0	1704.1	5322.1	50687.1
Environmental Control	60.0	-	45.0	5.0	110.0	5	18.0	90.0	5.0	25.0	225.0
Sub-Total Allocated	57759.6	26270.2	22333.8	7729.5	114093.1	5	13701.1	68505.5	11003.9	25880.7	208479.3
Sub-Total Non-Allocated (3) to Subsystems	2556.0	-	-	-	2556.0	5	200.0	1000.00	400.0	2000.0	5556.0
Payload Total (4)	$60315.6	$26270.2	$22333.8	$7729.5	$116649.1	5	$13901.1	$69505.5	$11409.9	$27880.7	$214035.3

(1) Includes adapter, no shroud.
(3) Excludes mission software.
(4) Excludes prime contractor's fee & NASA prog. mgmt.

(2) Equivalent of one unit with 2-year mission ops. & 5 units with launch ops. & support.

Fig. 3-27

Therefore, a corrected baseline SRS configuration has been created. The major differences in configuration between the corrected baseline SRS and the initial baseline SRS are:

- Solar array area was doubled
- Additional data handling was provided
- Active stabilization (spin-axis) control was provided

The reliability numbers and subsystem weights for the revised baseline SRS are shown in Fig. 3-28. Revised baseline cost estimates are shown in Fig. 3-29.

3.4 LAUNCH VEHICLE DATA AND INTERFACES

Characteristics of the Shuttle and candidate low-cost expendable launch vehicles which affected payload design (primarily structural loads and cargo bay environment) were provided by the Aerospace Corporation for this study, and are described in paragraph 3.1.6. In general, neither the performance capability nor the environmental conditions imposed burdensome constraints on low-cost payload design or operational modes for either the shuttle or expendable launch options. For the low-cost SEO, some growth in tug capability was found to be desirable to take full advantage of low-cost techniques and payload return capability when using the shuttle.

3.4.1 Launch Vehicle Environments

The shuttle cargo bay flight environments are not firmly established yet, but some current data is compared to that for the expendable Titan in Fig. 3-30, and the effects discussed in sub-section 8.1 Considerable mitigation of the external shuttle acoustics may be obtained by covering the inside of the cargo bay doors with sound attenuating material; this same treatment will serve to reduce thermal radiation from the inside of these doors, thus lessening the effect of ascent and reentry heating on the payload. With the shuttle, the transients due to staging are minimized and the high, steep-

Subsystem	Corrected Baseline SRS Reliability	Revised Baseline SRS Wt. (lb) or (kg)
Structure & Mechanisms	.9985	40.3 (18.3)
Environmental Control	.9994	6.4 (2.9)
Communications, Data Processing & Instrumentation	.952	61.4 (27.9)
Electrical Power	.9077	99.8 (45.3)
Stabilization & Control	.987	26.3 (11.9)
Propulsion & Attitude Control	.9139	27.0 (12.3)
Experiment Installation	.8247	55.7 (25.3)
Payload Total	.6408	316.9 (144.0)

Fig. 3-28 Corrected SRS Reliability and Weight Estimates

SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS
(1970 $ THOUSANDS)

PAYLOAD: SRS (Revised) LAUNCH VEHICLE: Baseline FLIGHT DURATION: 0.5 Yrs.

SUBSYSTEM	COSTS 1970 $ THOUSANDS			
	RDT&E Cost	Avg. Unit Cost	Avg. Opern's.	Total Cost *
Adapter	500	130	2	1028
Experiments	360	80	110	1120
Structures & Mechanisms	1280	120	11	1804
Electrical & Pyrotechnics	3950	700	11	6794
Guidance, Navigation, Stabilization & Control	2200	300	11	3444
Propulsion & Attitude Control	920	70	17	1268
Telemetry, Tracking & Command (incl. Instrumentation, Data Proces., Communications)	3000	600	33	5532
Environmental Control	440	60	-	680
Non-Allocated Costs	10	-	2	18
Payload Total	$12660	$2060	$ 197	$21688

* Includes 4 spacecraft and 4 operations.
** Prime contractor fee not included.

Fig. 3-29

LMSC-A990556

PAYLOAD ENVIRONMENT CHARACTERISTIC	LOW-COST EXPENDABLES		SPACE SHUTTLE		
	3-SEG. SRM/TITAN CORE II	TITAN III-L2	ASCENT	REENTRY	LANDING
Acceleration (g)					
Axial	+8.2	+5.4	+3.3	-0.5	-1.3
Yaw	±3.6	±2.4	±1.0	-2.0	-2.7, +1.0
Pitch	±3.5	±2.4	±1.0	±1.0	±1.0
Vibration (g^2/Hz) max. (Acceleration spectral density)	0.20	0.12	0.02	-	-
Temperature, °F (°K)					
Prelaunch (Conditioned)	80 (300)	80 (300)	80 (300)		
Ascent (Inside wall)	150 (340)	150 (340)	+150 (340) to -100 (200)		
On-Orbit	-	-	+150 (340) to -100 (200)		
Acoustic (OASPL, DB) (External)	145	146	158.5		

Fig. 3-30 Launch Vehicle Environment Comparison

fronted pyrotechnic shock that usually accompanies payload fairing separation is eliminated. Prelaunch thermal conditioning of payloads is easily accomplished, if necessary, for either launch mode, and is therefore not a significant tradeoff factor. The combination of a cryogenic stage and a payload under a common fairing, very probable if the Titan IIID/Centaur were considered, provides the same "cold wall radiation" effect and purge requirements as a shuttle payload might experience. In general, the shuttle payloads may experience a more severe acoustic environment but a milder shock, vibration, and acceleration environment and more easily accommodate mitigation measures than an expendable launch.

3.4.2 Launch Vehicle Interface Constraints

The constraints placed upon the payload by the launch vehicle have been inspected. This included review of the preliminary Interface Control Document for the Shuttle being coordinated by NASA/MSC. In general, no major technical problems seem to exist, but a considerable amount of detail analysis and design will be required to implement compatible interfaces. A few of the areas which were cursorily investigated are listed following; some require further study in follow-on effort.

3.4.2.1 Volume. The large 15x60 foot (4.6x18.3m) cargo compartment of the shuttle permits considerable freedom for handling a large variety of payloads, payload mixes, and sortie payloads without complex, specialized support structures for each different flight. It also provides operational modes not possible with expendable systems, and simpler interfaces.

3.4.2.2 Electrical Power. The on-board checkout and control functions, discussed in section 8.2, require more power, an effective interface with the shuttle data bus, and some crew participation.

3.4.2.3 Man-Safety. The manned shuttle imposes additional safety considerations not usually required for unmanned launches; however, the low-cost design

philosophy that reduces weight and volume constraints allows use of large safety factors for both structures and pressure vessels.

3.4.2.4 Abort. Shuttle abort modes require payload adaptability to propellant dump, pressure reductions and other safety measures, and a self-safing requirement is imposed for the payload recovery mode.

3.4.2.5 Contaminants. Contamination control is a problem with a vehicle such as the shuttle which is reused many times and the cargo bay is open much of the time on the ground. This problem is further complicated when there are a multiplicity of individual payloads making up the cargo. Solutions may not be entirely common with all payloads but are considered to be simpler with relaxed weight allowances (so that protective coverings can be used on critical payloads).

3.4.2.6 Ground Handling and Launching. Special consideration should be given to the payload installation or landing. When an expendable launch vehicle is used, the mating of the payload to the booster is usually done at the launch pad as one of the final steps in the launch vehicle build-up. Interfaces are verified and joint flight acceptance combined tests or simulated flights are performed as a prerequisite for initiation of final servicing, countdown and launch. If trouble develops in the payload it can usually be replaced without removing the launch vehicle from the launch stand. Shuttle payloads, on the other hand, are installed in the protected environment of the shuttle maintenance facility, or with the shuttle in the horizontal attitude. Handling, checkout, repair or adjustment, alignment and compatibility testing is probably much easier in this environment. These events start 15 or 16 work shifts before launch, are completed in five or six 8-hour shifts, and launch operations from then on are virtually independent of the payloads, except for automatic status monitoring and aliveness verification tests. Should difficulty develop with a low-cost payload, limited access within the cargo bay is possible for module replacement.

3.4.2.7 Special Propulsion Stages. Some payloads require propulsion stages, or tugs, to place them in the desired orbits. This introduces the additional payload/tug interfaces, as well as the tug/shuttle interfaces. The mechanical interfaces are straightforward, and are discussed in section 8.3.

3.4.2.8 Transportation System Capability. Because so many of the payloads in the mission model are destined for synchronous orbit, the Space Tug performance is particularly important. Fig. 3-31 shows the effect of optimizing the tug for syneq round trip missions. Comparative data is shown graphically in Fig. 3-32, illustrating the increased capability obtainable. Because the low-cost SEO is essentially saturating the capability of the 34.5 ft (10.5m) tug, further study of increasing the tug length and capability appears desirable so that single "round trip" missions can be flown to syneq orbit for SEO type payload replacement.

3.4.2.9 Propellant Loading, Vent, Dump. As long as the tug and shuttle propellants are compatible, several options exist for loading, transfer or dumping for abort, and refueling on orbit. If the tug is not cross-plumbed to the orbiter, safety considerations will probably dictate the ability to rapidly dump the tug propellants to reduce hazards and landing loads. One solution is to provide fill/dump couplings as part of the shuttle cargo bay doors or door access ports; after payload installation the tug is connected by short internal umbilicals to these couplings. Pressurants and purge gas are carried in the cargo bay to expel propellants and provide a safe, inert nitrogen atmosphere in the bay during this phase of a shuttle abort.

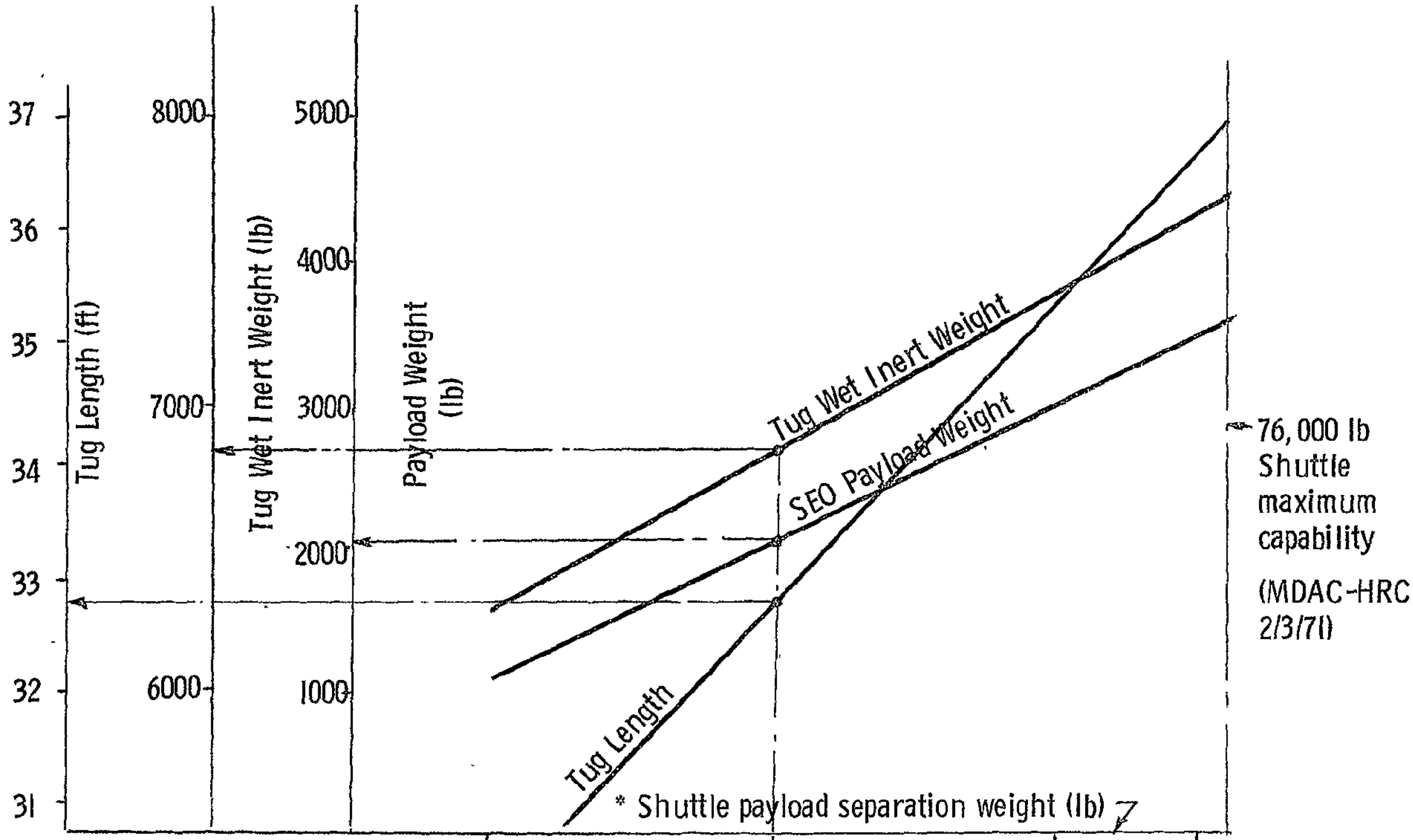
3.4.2.10 Multiple Payload Checkout. Tug checkout may be accomplished in the same manner as payload checkout, and may even use the same payload test set (section 8.2), perhaps slightly expanded if different telemetry systems and frequencies are employed. Pre-deployment checkout sequences need further study to determine time allocations and priorities; for example, should the tug or the payload be checked out first in an SEO mission? Obviously, if either is faulty after ascent the shuttle makes salvage possible by its

	W_p	W_i *	W_{TUG}	W_{PL}	W_{SEP}	L
	Tug Propell. (lb)	Tug Inert (lb)	Tug Total (lb)	SEO Total (lb)	Payload Separ. (lb)	Length Tug (ft)
Nominal Aerospace	57,760	6,840	63,600	2,360	65,960	34.5
Nominal LMSC (equiv.)	56,552	7,214	63,766	2,700	66,466	34.5
Optimized for SYNEQ (LMSC)	64,663	7,727	72,390	3,610	76,000	37.0

* Includes residuals and RCS

1 lb = .4536 kg
1 ft = .3048 m

Fig. 3-31 Tug Performance for Syneq Round Trip



* At 100 NM, 28.5^0
** LMSC data from STAR computation.

1 ft = .3048 m
1 lb = .4536 kg
1 nm = 1.852 km

Fig. 3-32 Comparative Data for Syneq Round Trip**

ability to return the cargo to earth. Some repair by modular replacement is possible for the low cost SEO, but perhaps the same modular concepts could be applied to tug design.

3.4.2.11 <u>Special Thermal Protection.</u> The same considerations of thermal radiation and acoustic damping apply to various payloads with perhaps additional thermal barriers required between tug (cryogenic propellants) and payload if long pad hold and/or coast times are necessary; such provisions could also be necessary within the payload fairing for an expendable launch system.

Section 4
PARAMETRIC COST OPTIMIZATION OF PAYLOAD SYSTEMS

Estimates of payload cost reductions that could be anticipated from the introduction of new launch systems were needed soon after the initiation of this Study. Aerospace, as the Fleet Analysis contractor, needed estimates of cost reductions, together with new weight and volume estimates as inputs to their Capture Analysis. Since design studies would not be completed until later in the Study, a parametric cost optimization analysis was conducted using the baseline payloads for starting values, thereby providing the required data for use by Aerospace in the preparation of their Interim Report.

4.1 COST OPTIMIZATION APPROACH AND METHODOLOGY

Proposed new launch vehicle systems introduce drastic changes in design and operational constraints for space payloads, including:

(1) greatly relaxed weight and volume restraints
(2) new capability for orbit revisit and refurbishment
(3) decreased cost and increased reliability of launch systems

The benefits from (1) are relatively direct, but must be realized by distributing them between subsystems in an approximately optimal manner. Those from (2) and (3) involve total program considerations and can only be fully realized by restating payload design requirements and constraints in terms, for example, of the changes in operational philosophy required to fully exploit Shuttle launch.

4.1.1 Cost Optimization Approach

The cost optimization methodology adopted uses as a baseline and point of departure historical data on actual payloads of conventional design. By means

of parametric mathematical models these are then reoptimized to respond to the new design and operational environment. These models are modular, so that specific subsystem models can either be adjusted by changes in parameters or replaced in toto as improved data becomes available.

The optimization criterion chosen is that of minimum program cost maintaining program objectives, including program duration, mission data requirements and mission performance requirements, constant. The number of launches required during the program is, however, optimized.

Optimization is by a general optimization computer program called POP, derived from the SWORD computer program, which minimizes program cost for each payload and launch vehicle combination by appropriate allocation of weight, cost and reliability among subsystems, using the parametric model of the payload to identify the most desirable allocation of these resources.

4.1.2 Payload Optimization Computer Program

The Payload Optimization Program (POP) was employed to minimize the total program cost by finding the optimum values of the weight, reliability and component reliability (or quality) of each subsystem subject to a constraint on total payload weight and subject to a required program operating time and, as appropriate to certain space shuttle-launched missions, mission time between ground refurbishments or on-orbit maintenance. Thus, POP served as the analytical tool for integrating and evaluating on a standard basis all the input baseline and asymptotic data on costs, weights and reliabilities.

4.1.2.1 Gradient Optimization Method. SWORD and POP utilize an iterative step-by-step optimization scheme. At each step, all the independent variables (i.e., the subsystem weights, reliabilities and component reliabilities) are incremented in an attempt to improve the payoff quantity (total program cost) and to hold the constrained variable(s) (the total payload weight) constant. The mathematical principle behind each step is expressed as follows:

$$\delta\bar{X} = \bar{\eta} \left(K_F \frac{\partial f}{\partial \bar{X}} + \bar{K}_W \frac{\partial \bar{w}}{\partial \bar{X}} \right)$$

where $\bar{X}$ = the vector of all the independent variables

$\bar{\eta}$ = a vector of weighting factors which act as scaling factors and convergence adjusters

f = payoff quantity

$\bar{w}$ = vector of (all) the constrained variable(s)

K_F, $\bar{K}_W$ = variable coefficients which set the size of each incremental change and which determine the relative improvement in f and the correction(s) in $\bar{w}$.

The iterations cease when Δf is acceptably small or when the maximum allowable number of steps has been taken, this number being selected by the user for each computer run.

After POP was coded and in operating condition, about a month was spent in testing and revising the models, comparing preliminary results with practical design experience, trying gain-setting methods and testing the program's convergence. A simple effective rule was developed for initially setting the gains. The program's convergence was tested by attempting to find a solution to extremely different cases from entirely different starting values of the independent variables. The following table summarizes the degree of agreement between the converged solutions:

Item	Differences in Initial Values	Agreements Between Converged Solutions
Total program cost	-	0.05 - 0.3%
Subsystem weight	19 - 320%	0 - 3 %
Subsystem reliability	3 - 22%	0.16 - 0.9%
Component reliability	7 - 80%	0.2 - 5 %

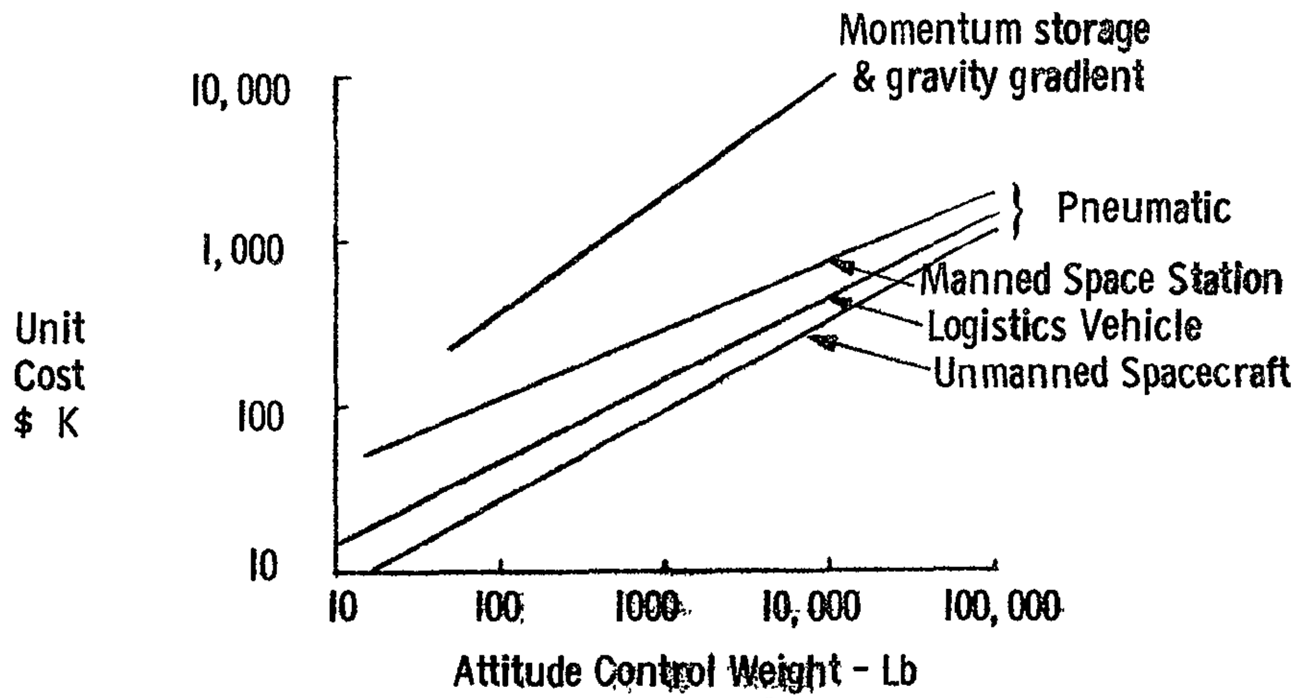
Not only does this table describe the quality of convergence but it also demonstrates that (1) for these examples, there do not appear to be any local minima near the optimum solutions that could cause spurious results, and (2) the variation in the value of the payoff quantity (total program cost) is about an order of magnitude smaller than the disagreements in the independent variables; that is, small changes in the independent variables have little influence on the optimum payoff value.

4.1.3 Model Algebra - Subsystem Models

The subsystem models consist of algebraic expressions defining the RDT&E and unit costs of each subsystem "j" as functions of the subsystem weight and reliability. Optionally, tabulated data could be used instead of algebraic expressions. Operations costs were not so expressed in the present analyses but were included as constant, non-tradeable terms in the total cost. Each model is a sub-routine of the main program and can readily be replaced wholly or in part by an improved model. Additionally, each such model can be adjusted to represent specific cases by appropriate selection of parametric coefficients.

Two dimensional relationships are first considered below, between costs and subsystem weight, unit reliability and redundancy in turn. These are then combined into single, multi-dimensional expressions.

4.1.3.1 Subsystem Cost vs Weight. For this study it was necessary to develop cost vs weight relationships that reflected the frequently-expressed belief that increasing the allowable weight (for any particular subsystem) would have the effect of decreasing the total cost of that subsystem. Traditional cost vs weight CERs exhibit the opposite trend as shown in Fig. 4-1, with costs increasing with increasing weight. Such CERs, while valuable in deriving planning estimates for costs of spacecraft, have been developed utilizing historical data from a large number of missions whose objectives (and hence subsystem requirements) have varied greatly, one from another. The increasing weight, in many cases, reflects increasing capability as well as increasing costs.



1 lb = .4536 kg

Fig. 4-1 - Traditional Cost vs Weight CER

Hypothetical cost vs weight relationships for subsystems wherein the functional requirements have been held constant are shown in Fig. 4-2 . Somewhat the same characteristics are displayed by all those shown; as the weight limits are lowered, the costs necessarily increase because of the need to miniaturize components, etc. On the other hand as the weight increases further, the costs ultimately begin to rise again due, as much as anything else, to the cost per pound of the materials. For different subsystems it is expected that the specific shapes will vary, but the characteristics of the curves are similar from one subsystem to another.

While it was not possible to develop cost vs weight relationships that extend over the full range of potential weights, it was possible to approximate such curves for relatively small increases in subsystem weight. In Fig. 4-3 are demonstrated the steps required, beginning with the cost and weight of the baseline payload subsystem, estimating the asymptotes depicting the minimum weight regardless of cost and the minimum cost regardless of weight respectively, and constructing a hyperbola through the baseline point and asymptotic to the two lines. The resulting relationship, while not exact, permits an approximation of the effects of increasing weight while maintaining functional requirements a constant.

It is mechanized in the following form for unit costs C_{uj} of subsystem "j" and in a precisely parallel form for RDT&E costs C_{rj}

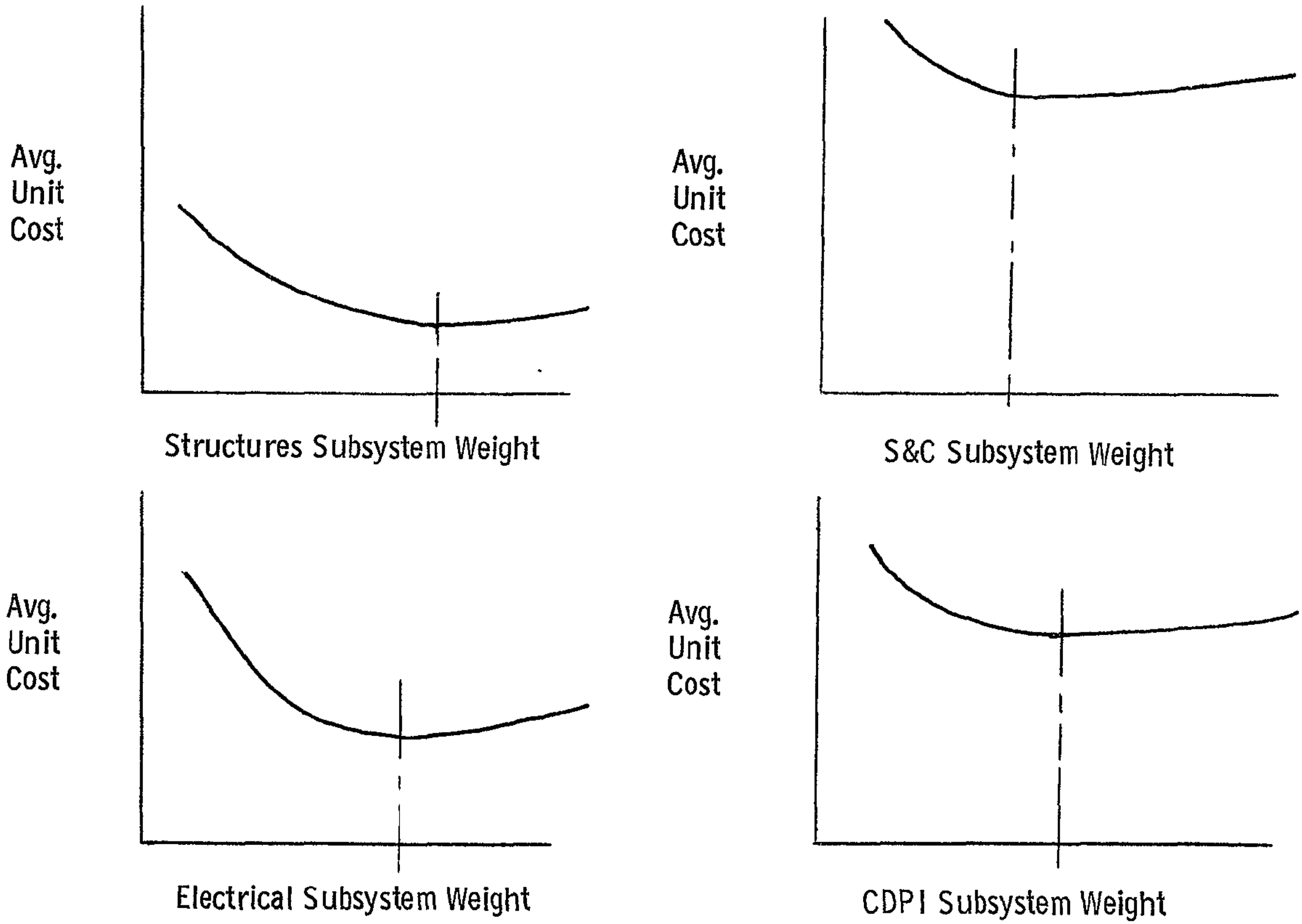
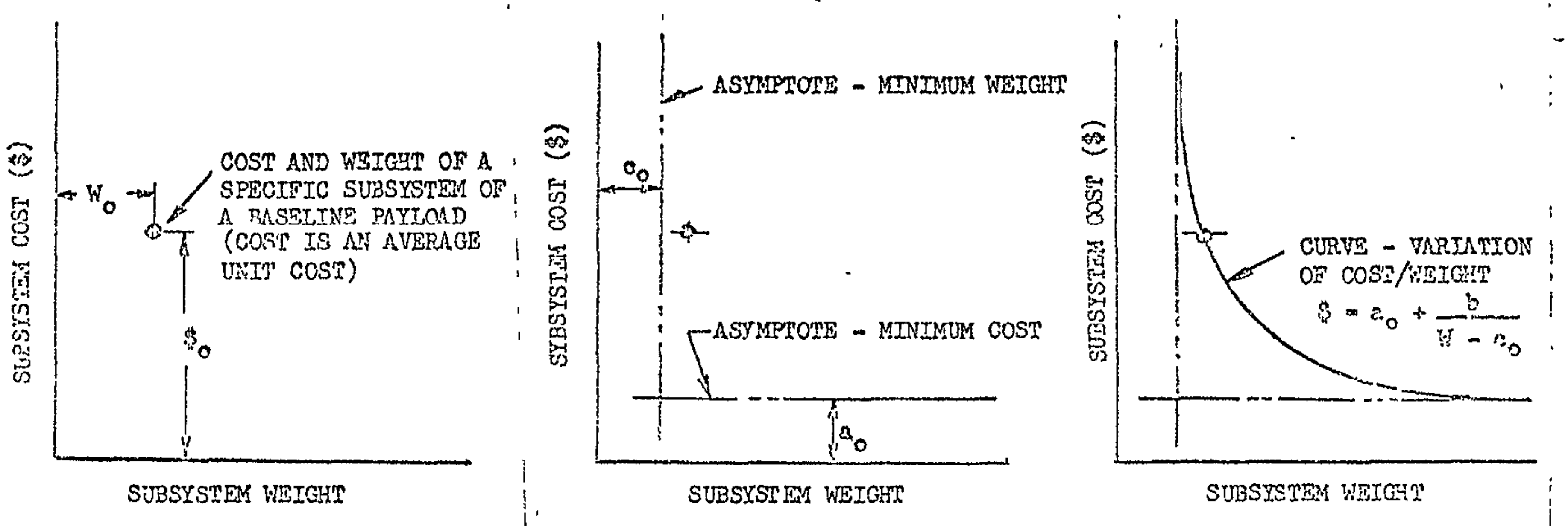


Fig. 4-2 Constant-Capability Subsystems - Hypothetical Cost vs Weight CERs



A

IDENTIFY THE APPORTIONED COST OF A SUBSYSTEM (PORTION OF TOTAL PROGRAM COST DIVIDED BY QUANTITY OF FLIGHT ARTICLES)

B

BY ANALYSIS AND INSPECTION OF PARTICULAR SUBSYSTEM REQUIREMENTS, DETERMINE THE ASYMPTOTES:

(1) LOWEST COST, DISREGARDING WEIGHT

(2) LOWEST COST, DISREGARDING COST

C

CONSTRUCT AN HYPERBOLA THROUGH THE BASELINE DATA POINT AND THE TWO ASYMPTOTES.

THIS REPRESENTS THE APPROXIMATE RELATIONSHIPS OF WEIGHT AND COST FOR OTHER DESIGNS OF THE SUBSYSTEM, ASSUMING THAT THE FUNCTIONAL CAPABILITY OF THE SUBSYSTEM REMAINS CONSTANT

Fig. 4-3 DEVELOPMENT OF A TYPICAL SUBSYSTEM COST VERSUS WEIGHT MODEL

$$C_{uj} = C_{ujb} \left\{ 1 + \frac{(Cujb/Cujm - 1)\ (Wjb - Wjm)}{Wj - Wjm} \right\} \qquad (4\text{-}1)$$

C_{ujb} = unit cost of baseline subsystem "j"

C_{ujm} = minimum unit cost regardless of weight

W_j = weight of subsystem "j"

W_{jb} = weight of baseline subsystem "j"

W_{jn} = minimum weight regardless of cost

The parallel form for subsystem RDT&E costs is, of course,

$$C_{rj} = C_{rjb} \left\{ 1 + \frac{(C_{rjb}/C_{rjm} - 1)\ (W_{jb} - W_{jm})}{W_j - W_{jm}} \right\} \qquad (4\text{-}2)$$

In the first iteration, the baseline values for subsystem costs and weight are those derived from the historical data from actual payloads. In later iterations, however, one may substitute data derived from other payload preliminary design data.

4.1.3.2 Subsystem Cost vs Reliability. Subsystem reliability can be varied by varying unit/component quality or by varying redundancy, the optimum choice depending on the constraints imposed on the system. For example, if low payload weight is essential, high component reliability is generally preferable to redundancy as a means of achieving reliability. On the other hand, if low cost is a primary objective and weight is not critical the reverse may be true, if practically possible. It is therefore important to represent both mechanisms in the model.

4.1.3.3 Unit Quality. There is evidence to support models for both RDT&E and unit costs of the form (for a fixed nominal mission duration).

$$\text{cost} \propto \text{MTBF}^{\beta} \propto \left\{ - \ln\ (\text{reliability}) \right\}^{-\beta}$$

Specifically, the relationship has been applied as follows:

unit cost $$C_{uj} = C_{ujb} \left(\frac{\ln R_{cjb}}{\ln R_{cj}} \right)^{\beta_u} \tag{4-3}$$

RDT&E cost $$C_{rj} = C_{rjb} \left(\frac{\ln R_{cjb}}{\ln R_{cj}} \right)^{\beta_r} \tag{4-4}$$

where R_{cj} = "unit/component reliability" in subsystem "j"

R_{cjb} = corresponding reliability in baseline subsystem

The physical quantity involved here is the ratio of the failure rate of the proposed subsystem to that of the baseline subsystem, the quantity R_{cj} being of the nature of a dummy variable to represent the failure rate of a system with the same redundancy as the baseline system but differing component quality. For convenience in later combination of the expression for the effects of component quality and of redundancy, the above quantities were actually defined in the model as follows:

R_{cjb} = reliability of baseline subsystem (= R_{jb})

R_{cj} = reliability of subsystem with the same redundancy as the baseline system but differing component quality

4.1.3.4 <u>Subsystem Cost and Weight vs Redundancy</u>. Much dedicated expertise has been devoted, successfully, to develop methodologies by which to optimize the redundancy logic of complex subsystems for cost, weight or some combination of these. These methodologies are highly effective in their specific applications. The present need, particularly for the first iteration of payload reoptimization to the new environment, is, however, for rather general models which will generate approximate subsystems requirements and goals for the totally new systems operational environment and will effectively focus detailed analyses in the right areas. The models proposed below therefore emphasize ease of application,

subject to correct identification of the trends and the optimum areas to direct more detailed analyses.

4.1.3.5 Redundancy by Subsystem. The reliability relationship for n subsystems in parallel can be expressed in the form

$$1 - R(n) = \{1 - R(1)\}^{n} \tag{4-5}$$

Standby redundancy, which is more efficient, is expressible in the somewhat more inconvenient form

$$R(n) = R(1) \sum_{j=1}^{n} \frac{(-\ln R(1))^{n-1}}{(n-1)!} \tag{4-6}$$

It is desirable to avoid having a zero redundancy case as a cost reference in this application since the baseline payload will in general incorporate redundancy. The zero redundancy case is used in the following formula.

$$\frac{C_{uj}}{C_{ujb}} = \frac{W_j}{W_{jb}} = \frac{n}{n_{baseline}}$$

$$= 1 + \frac{\ln (1 - R_j) - \ln (1 - R_{jb})}{\ln(1-R_{jb}) + \alpha_1 \ln (1 - R_{jbl})} \tag{4-7}$$

where R_{jbl} = reliability of baseline subsystem with no redundancy. Since this expression is insensitive to the term in R_{jbl} this need not be determined very precisely.

On the assumption that subsystem cost and weight are proportional to the number of replicate subsystems the above expression is exact, with α equal to zero, for parallel redundancy and is a good approximation, with $\alpha_1 = 0.2$, for standby redundancy. With α equal to 0.1 it is a good approximation to either, or to a mixed type of redundancy.

In general only part of RDT&E costs may be expected to be proportional to redundancy. RDT&E costs were therefore modelled assuming that a proportion "α_3" of them would follow a relationship similar to that for unit costs, so that

$$\frac{C_{rj}}{C_{rjb}} = (1 - \alpha_3) + \alpha_3 \left\{ 1 + \frac{\ln (1-R_j) - \ln (1-R_{jb})}{\ln (1-R_{jb}) + \alpha_1 \ln (1-R_{jbl})} \right\} \quad (4\text{-}8)$$

The value of α_3 was estimated on a case by case basis and was generally in the range of 0.5 - 0.7.

4.1.3.6 Redundancy by Component. In addition to the above, an alternative expression has been developed for redundancy by component which, however was not used in the initial optimization runs. This was of the form

$$\frac{\text{Cost}}{\text{Baseline Cost}} = \frac{\text{Weight}}{\text{Baseline Weight}} = \frac{\ln N_j - \ln (- \ln R_j)}{\ln N_j - \ln (- \ln R_{jb})} \quad (4\text{-}9)$$

where N_j = number of components in the subsystem

4.1.3.7 Operations Costs of Subsystems. The structure of the parametric model allows the option of modelling operations costs by subsystem as functions of, for example, subsystem reliability. In the present analyses, however, this option was not exercised. Operations costs were included in the total program cost tradeoff but as an invariant item.

4.1.3.8 Combination of Two-Dimensional Relationships. RDT&E and unit costs have been modelled above as functions of:

- subsystem weight
- subsystem reliability, varying unit/component reliability at constant redundancy
- subsystem reliability, varying redundancy at constant unit/component reliability

It is now necessary to combine these in a rational manner. In doing so, two interactions between the two-dimensional relationships have been recognized:

(1) Changes in unit/component reliability change the reliability of the baseline system to which the cost vs "reliability by redundancy" model is applied.

(2) Changes in redundancy change the weight of the baseline system to which the cost vs weight model is applied.

Item (1) has been expressed in the manner shown below in which the function R_u is the ratio of the cost and weight of the proposed subsystem "j" to those of the baseline subsystem "j". After the baseline costs have been adjusted for changes in unit/component reliability. This expression, which defines a weight ratio as well as a cost ratio, is then incorporated into the weight/cost model, resulting in the total ("four-dimensional") model. The actual subsystem weight W_j, is, in effect, rescaled for the changes in redundancy in order to place it correctly on the cost vs weight curve. Parallel relationships for RDT&E costs are shown below.

It will be noted that the models are such that segments of the model, such as the redundancy models "R_u" and "R_r", can readily be changed if more detailed analyses should so require.

Unit cost C_{uj} is expressed as a function of subsystem weight W_j, unit reliability (R_{cj}) and subsystem reliability (R_j):

$$C_{uj} = C_{ujm} \underbrace{\left(\frac{\ln R_{cjb}}{\ln R_{cj}}\right)^{\beta u}}_{\text{UNIT RELIABILITY (QUALITY)}} \underbrace{Ru}_{\text{REDUNDANCY}} \underbrace{\left\{1 + \frac{(C_{ujb}/C_{ujm} - 1)(W_{jb} - W_{jm})}{W_j/R_u - W_{jm}}\right\}}_{\text{WEIGHT}}$$

where $R_u = 1 + \dfrac{\ln(1 - R_j) - \ln(1 - R_{jb}\, R_{cj}/R_{cjb})}{\ln(1 - R_{jb} R_{cj}/R_{cjb}) + \alpha_1 \ln(1 - R_{jb_1} R_{cj}/R_{cjb})}$

and C_{ujb}, W_{jb}, R_{cjb} and R_{jb} are baseline values (inputs)
C_{ujm}, W_{jn} "are minimum possible values" (inputs)
R_{jb_1} is reliability of baseline subsystem without redundancy (input)

RDT&E costs (C_{rj}) are expressed as a function of subsystem weight (W_j), unit reliability (R_{cj}) and subsystem reliability (R_j):

$$C_{rj} = C_{rjm} \underbrace{\left(\frac{\ln R_{cjb}}{\ln R_{cj}}\right)^{\beta_r}}_{\substack{\text{UNIT RELIABILITY} \\ \text{(QUALITY)}}} \underset{\substack{\uparrow \\ \text{REDUNDANCY}}}{R_r} \underbrace{\left\{1 + \frac{(C_{rjb}/C_{rjm} - 1)(W_{jb} - W_{jm})}{W_j/R_r - W_{jn}}\right\}}_{\text{WEIGHT}}$$

where $R_T = 1 - \alpha_3 + \alpha_3 \quad 1 + \dfrac{\ln(1 - R_j) - \ln(1 - R_{jb}\, R_{cj}/R_{cjb})}{\ln(1 - R_{jb}\, R_{cj}/R_{cjb}) + \alpha_1 \ln(1 - R_{jb_1}\, R_{cj}/R_{cjb})}$

and C_{ujb}, W_{jb}, R_{cjb} and R_{jb} are baseline values (inputs)

C_{ujm}, W_{jn} are "minimum possible values" (inputs)

R_{jb_1} is reliability of baseline subsystem without redundancy

4.1.4 <u>Model Algebra-System Model</u>

The total payload is modelled as the sum of the subsystems, with provisions for a residual "subsystem" to take care of costs not allocatable to hardware subsystems. The conventional subsystems can be subdivided if desired for modelling convenience if the conventional breakout associates components with very different characteristics from the point of view of the model.

Each subsystem is defined to the system by the following:

subsystem RDT&E costs	C_{rj}
subsystem unit cost	C_{uj}
subsystem operations cost	C_{opsj}
subsystem reliability	R_j
subsystem weight	W_j

The system is then defined to the optimization criterion by the following:

payload RDT&E cost	$C_r = \Sigma C_{rj}$
payload unit cost	$C_u = \Sigma C_{uj}$
payload operations cost	$C_{ops} = \Sigma C_{opsj}$
payload weight	$W = \Sigma W_j$
payload reliability	$R = \Pi R_j$

4.1.5 Optimization Criterion

The computer program allocates weight, reliability and cost among subsystems and optimizes the payload against the launch vehicle interface according to a defined optimization criterion. The criterion presently selected is the expected total cost of the program required to execute the nominal task it is to perform.

The program cost "C_{PR}" is expressed as the sum of:

- expected launch vehicle costs for the program, allowing for launch vehicle and payload unreliability
- expected sum of payload unit costs, with maintenance and refurbishment where relevant
- payload RDT&E
- operations cost

A fundamental and critical quantity which is strongly dependent on the definition and requirements of the program is:

M_{SL} = expected number of successful missions per successful launch

The quantity M_{SL} has been expressed in terms of payload reliability, R, where the latter is the probability that the payload, having been successfully launched, will complete one nominal mission. In the case presently modelled credit is given for completing fractions of the nominal mission. Figure 4-4 shows how M_{SL} varies with payload reliability for two program times, where the latter are quoted in terms of the nominal mission time. It should be noted that it is entirely possible, if the mission is not terminated at the nominal time, for the quantity M_{SL} to be greater than unity.

An important practical case arises with limited expendables or if subsystems or components are expected to wear out, or experience an intolerable degree of performance degradation as a result of parametric drift prior to random (exponential) failure. An approximate representation of this case was provided by providing for truncation at a single, representative time equal to X_W nominal missions.

4.1.5.1 Total Program Cost Including On-Orbit Maintenance and/or Ground Refurbishment. This cost model applies to the operational mode in which a payload is maintained on orbit during the operating mission life of a set of experiments/sensors, is retrieved from orbit and returned to the ground for refurbishment and re-outfitting with a different set of experiments/sensors, and is then returned to orbit for the next mission period. There may be several cycles of refurbishment during the entire time of program performance. Each period between the initial launch and the first refurbishment, between refurbishments, or between the last refurbishment and the end of the flight

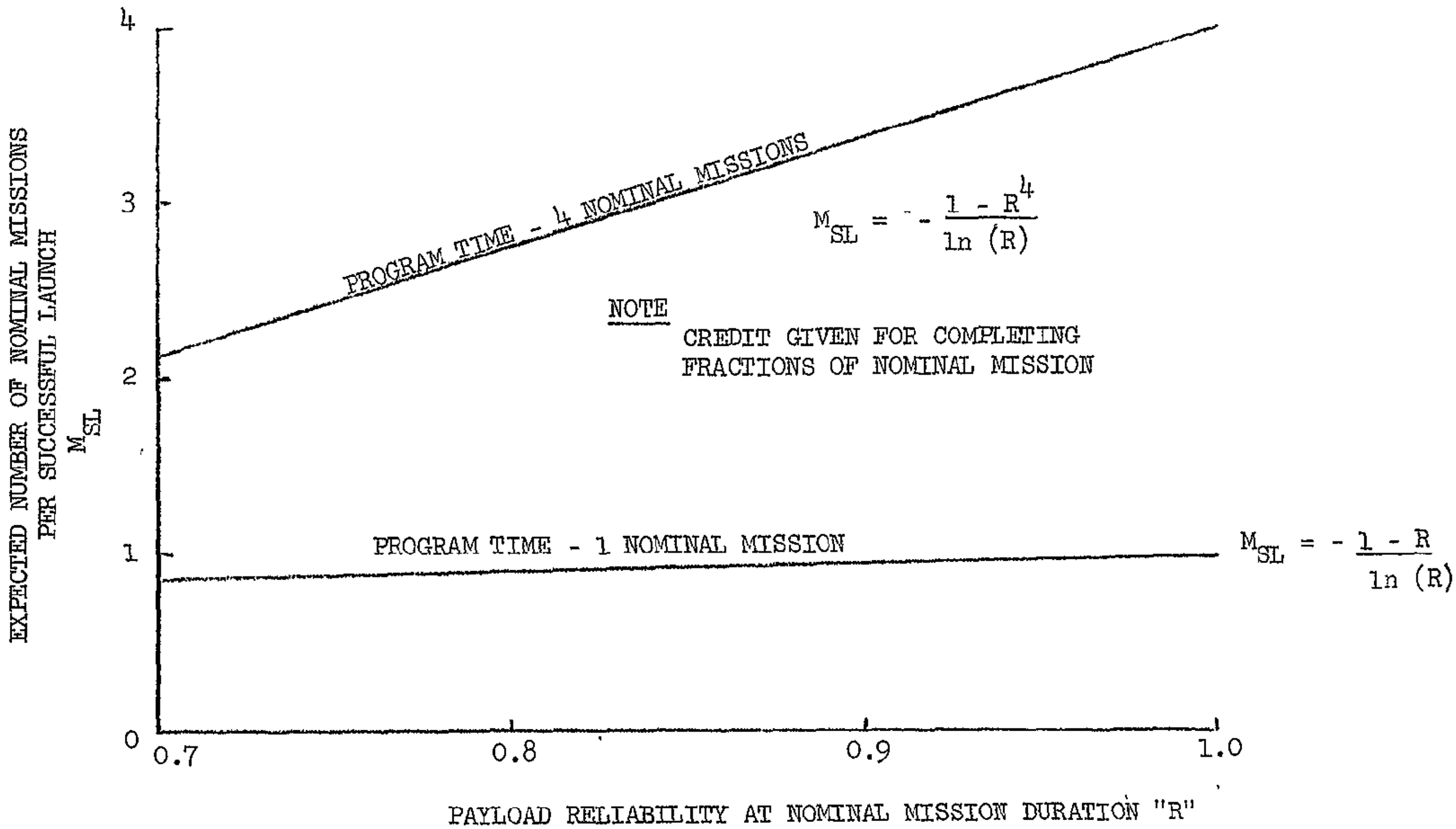


Fig. 4-4 Effect of Program Time on M_{SL}

program is termed a "mission block"; to simplify the parametric analysis, it was assumed that during a particular flight program all the mission blocks had the same duration.

The total program cost C_{PR} is found as follows:

$$C_{PR} = C_r + C_{u/MR} + T_P C_{OPS} + N_{LV} C_{LV}$$

where Cr = total RDT&E cost

$C_{u/MR}$ = total unit cost with orbital maintenance and refurbishment

T_p = duration of the program

C_{OPS} = total program operational cost per unit of time

N_{LV} = number of launches

C_{LV} = cost of each launch

The total unit cost is composed of the following terms:

$$C_{u/MR} = C_u + BC_{MT} + C_{REF}$$

where C_u = sum of the payload unit cost

B = number of mission blocks in the total flight program = T_F/T_M

T_M = duration of a mission block

C_{MT} = maintenance cost per mission block = $(T_M/M_{SL} - 1)\ R_{MT}C_u$

M_{SL} = expected payload life per successful launch

R_{MT} = ratio of maintenance to unit costs

C_{REF} = total refurbishment cost = $(B-1)\ R_{REF}\ C_u$

R_{REF} = ratio of refurbishment to unit costs

The number of launch vehicles is found by summing up the following requirements:

N_{LV} = B (one initial launch + number of maintenance launches per mission block)

+ number of launches to retrieve payload for refurbishments

$$= B \frac{1 + (T_M/M_{SL} - 1)}{P_L} + \frac{B - 1}{P_L}$$

$$= \frac{(T_P/M_{SL}) + B - 1}{P_L}$$

where P_L = expected probability of launch success.

4.1.5.2 Total Program Cost Without Maintenance or Refurbishment. In the case of the expendable payload, which is neither maintained nor refurbished, the expression for expected program cost becomes much less complex, as follows:

Expected program cost	$C_{PR} = C_r$	(RDT&E cost)
+	$C_u\ N_{LV}$	(payload unit costs)
+	$T_P\ C_{OPS}$	(operations costs)
+	$N_{LV}\ C_{LV}$	(launch vehicle costs)
where	N_{LV} =	number of launches required
	=	$N_M/(M_{SL}\ P_L)$

4.2 DEVELOPMENT OF COMPUTER INPUT DATA

The preceding section describes a general parametric model of the payload and of its interface with the launch vehicle. This section describes the application of this model to each payload/launch vehicle combination, the development of estimates of the model coefficients, and the use of techniques for modifying the analysis model so that it was compatible with design reality.

4.2.1 Evaluation of Cost-Reduction Potential

The first step in the determination of cost reduction potential in each of the baseline payloads was the general design analysis of each subsystem. Those characteristics of the hypothetical low-cost payloads showing cost reduction potential were identified and their influence upon payload program costs were qualitatively evaluated in terms of high effect, moderate effect, or low effect.

The next step involved the more detailed assessment of program costs in terms of the amount of cost reduction which could be obtained. To assure that all cost-reduction areas were being considered, a "checklist" type of matrix was prepared. All cost-reduction assumptions were listed and cross-correlation to the affected cost categories was identified. The matrix later was used as a reference in estimating the cost reductions possible in each of the cost categories.

4.2.2 Determination of Computer Input CERs

One of the basic inputs to the cost-optimization computer program is the weight-versus-cost Cost Estimating Relationships (CERs). The development of these CERs is described in the following paragraphs. The traditional CER is a curve which provides increase of cost with increase in weight of a subsystem. These historical curves were usually developed using data from various types of payloads wherein increase in subsystem weight was synonymous with increased complexity and performance capability of the subsystem; the cost of the more complex or higher capability subsystem was therefore higher also.

In the Payload Effects Study, the performance (and the complexity) of each subsystem has been assumed a constant. Any increase in weight of the subsystem will allow use of simpler design approaches, less-dense packaging, a decrease in intra-connect complexity, use of less-costly materials, and similar cost-reduction approaches. The cost of the subsystem will therefore decrease as the weight is increased.

4.2.2.1 Development of New Cost-Weight CER. A "standardized" CER for use with the POP computer program has been developed to cover this condition and is shown in Fig. 4-5. In the construction of this CER, the two asymptotes must be established and a hyperbola curve through the baseline cost/weight point and asymptotic to the two lines then represents the cost-weight relationship. The logic involved in the selection of the hyperbola and its limitations were described in par. 4.1.3.1 As shown, the horizontal asymptote represents the minimum cost of the subsystem if no constraint is placed upon weight. The vertical asymptote represents the minimum weight of the subsystem if no constraint is placed upon cost. The algebraic equivalent of this curve was provided to the cost-optimization computer program. Curves were developed separately for RDT&E cost vs weight and Unit Recurring Cost vs Weight. The actual shape of this curve was not known at this stage of the analysis. The assumed hyperbolic shape does, however, represent a convenient initial mechanization of the fact that decreasing cost returns will be experienced as weight is increased.

4.2.2.2 Estimation of the Minimum Weight Asymptotes. The minimum-weight asymptotes were estimated for the RDT&E and Unit Recurring cost of each subsystem, of each payload, in combination with a particular launch vehicle type. Since preliminary analyses had indicated that this asymptote was not particularly influential in forecasting low-cost estimates, a detailed analysis of minimum weights was not performed. Rather, approximate engineering estimates were made.

4.2.2.3 Estimation of the Minimum-Cost Asymptotes. Considerable care was exercised in deriving the minimum-cost asymptotes of the CERs. Each cost category within the RDT&E and Unit Recurring areas was individually inspected and the approximate amount of cost reduction estimated. All of the cost-reduction assumptions listed were applied. The cost reductions were summed by subsystem for each payload/launch vehicle combination; these in turn were subtracted from the corresponding baseline costs (with proper factors applied) to obtain the minimum-cost asymptotes. Figures 4-6 through 4-9 are tabulations of these derived CER asymptotes; these data were provided as inputs to the cost-optimization computer runs.

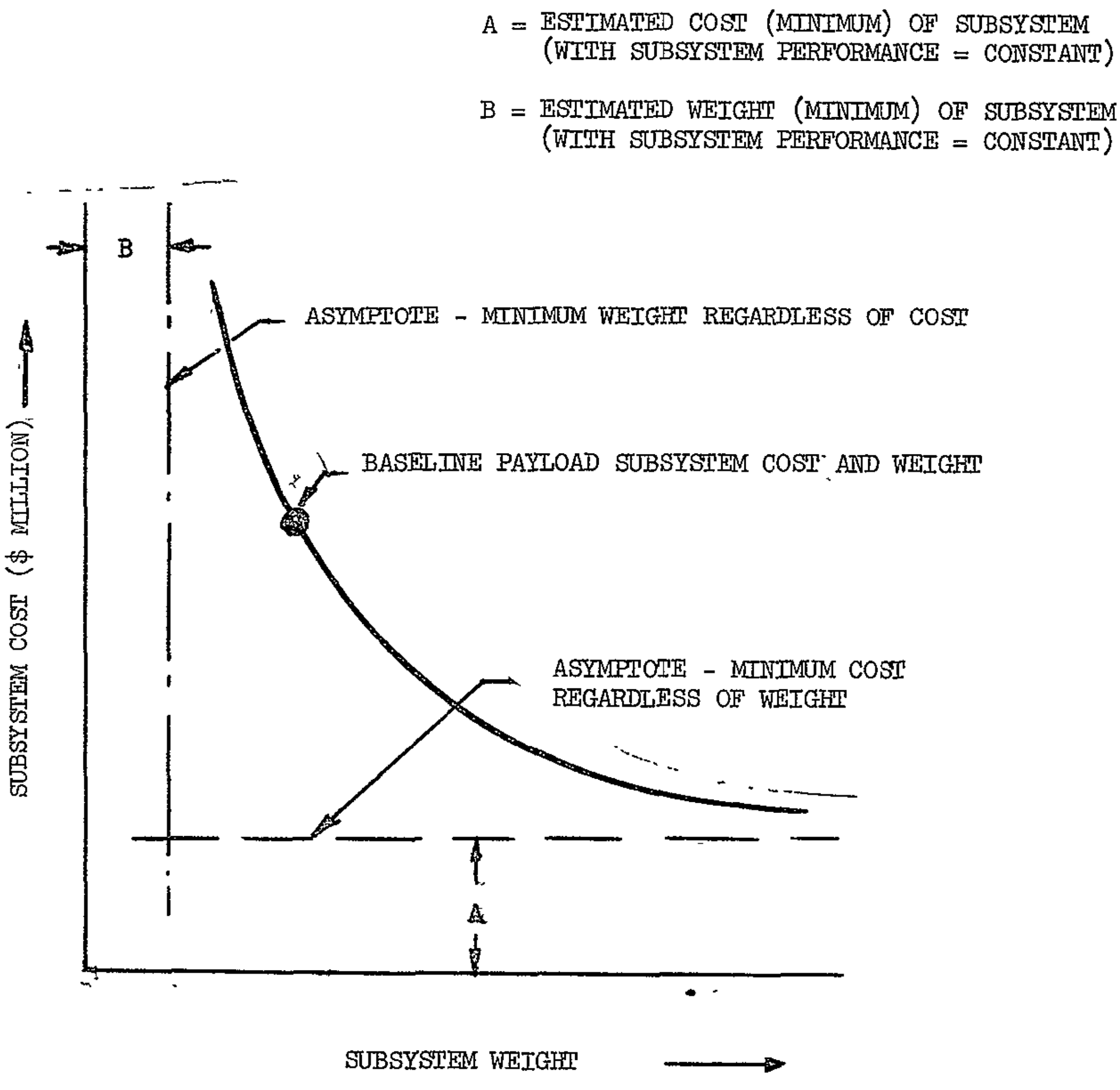


Fig. 4-5 Typical CER - Cost vs Weight

SUBSYSTEM \ LAUNCH* VEHICLE	RDT&E COST ($1000)			UNIT RECURRING COST ($1000)		
	ACE	LCE	SS	ACE	LCE	SS
EXPERIMENTS	300	300	176	47	47	47
STRUCTURES & MECHANISMS	600	600	390	24	24	24
ELECTRICAL & PYRO	2,450	1,650	1,200	250	250	189
STABILIZATION & CONTROL	70	70	27	16	16	16
PROPULSION / ATTITUDE CONTROL	275	275	275	50	50	50
CDPI	2,500	2,500	1,150	225	225	225
ENVIRONMENTAL CONTROL	150	150	75	22	22	22

* ACE = Alt. Current Expendable
LCE = Low-Cost Expendable
SS = Space Shuttle

Fig. 4-6 CER MINIMUM COST ASYMPTOTES - SRS

LAUNCH* VEHICLE / SUBSYSTEM	RDT&E COST ($1000)			UNIT RECURRING COST ($1000)		
	ACE	LCE	SS	ACE	LCE	SS
EXPERIMENTS	7,217	6,217	3,017	6,250	4,650	4,250
STRUCTURES & MECHANISMS	8,174	7,294	4,544	4,100	3,200	3,200
ELECTRICAL & PYRO	14,333	13,483	7,083	2,800	2,800	2,350
STABILIZATION & CONTROL	69,269	63,969	29,969	10,200	8,800	5,900
PROPULSION	-	-	-	-	-	-
ATTITUDE CONTROL	2,995	2,995	1,505	230	230	230
CDPI	37,923	30,123	14,623	4,230	4,100	2,680
ENVIRONMENTAL CONTROL	5,735	5,405	3,295	750	1,300	1,300

* ACE = Alt. Current Expendable
LCE = Low-Cost Expendable
SS = Space Shuttle

Fig. 4-7 CER MINIMUM-COST ASYMPTOTES - OAO(B)

SUBSYSTEM \ LAUNCH* VEHICLE	RDT&E COST ($1000)			UNIT RECURRING COST ($1000)		
	ACE	LCE	SS	ACE	LCE	SS
EXPERIMENTS	30,291	30,291	16,211	1,414	1,414	1,139
STRUCTURES & MECHANISMS	6,200	6,200	3,600	1,100	1,100	900
ELECTRICAL & PYRO	8,046	8,046	4,086	1,436	1,436	1,161
STABILIZATION & CONTROL	7,020	7,020	4,000	950	950	800
PROPULSION	-	-	-	-	-	-
ATTITUDE CONTROL	1,814	1,814	1,253	173	173	118
CDPI	19,901	19,901	10,771	2,071	2,071	1,521
ENVIRONMENTAL CONTROL	105	105	105	35	35	35

* ACE = Alt. Current Expendable
LCE = Low Cost Expendable
SS = Space Shuttle

Fig. 4-8 CER MINIMUM-COST ASYMPTOTES - SEO

SUBSYSTEM \ LAUNCH* VEHICLE	RDT&E COST ($1000)			UNIT RECURRING COST ($1000)		
	ACE	LCE	SS	ACE	LCE	SS
EXPERIMENTS	29,290	29,290	15,200	3,400	3,400	2,800
STRUCTURES & MECHANISMS	6,800	6,800	3,950	1,050	1,050	800
ELECTRICAL & PYRO	8,200	8,200	4,200	1,500	1,500	1,300
STABILIZATION & CONTROL	7,500	7,500	4,500	1,950	1,950	1,600
PROPULSION	3,500	3,500	3,000	250	250	220
ATTITUDE CONTROL	1,814	1,814	1,253	167	167	120
CDPI	20,000	20,000	10,900	2,800	2,800	2,100
ENVIRONMENTAL CONTROL	215	215	215	40	40	40

* ACE = Alt. Current Expendable
LCE = Low Cost Expendable
SS = Space Shuttle

Fig. 4-9 CER MINIMUM-COST ASYMPTOTES - MARS ORBITER

4.2.3 Model Modification for Design Practicability

As the parametric analyses progressed it was found necessary, as might be expected to make adjustments to and impose constraints on the model to represent limits imposed by consideration of design practicability.

4.2.3.1 Subdivision of Subsystems. When a subsystem, such as an experiment subsystem, contained packages with very different parametric characteristics, it was divided into two more homogeneous groups. For example, the OAO experiments package was divided into a primarily mechanical package (telescope, mirrors, etc.), and a primarily electronics group. Cost/Reliability tradeoffs were permitted for the electronics group but not for the mechanical group.

4.2.3.2 Failure Rate Ratio and Parametric Drift. For the CDPI and S&C subsystems in particular, in spite of the very conservative approach used of modelling cost vs reliability on the basis of redundancy at the subsystem level, the parametric analysis results called, in some cases, for high redundancy combined with low grade parts whose failure rates were 10-15 times those for normal space vehicle practice. Such failure rates correspond approximately to lower grade commercial parts and are subject to fairly rapid parametric drift. For the case of the OAO/Shuttle combination, the optimum average refurbishment frequency was about three times per year. In order to conform to this desired average life the failure rate ratio was therefore restrained so as not to exceed 5. This corresponds to the expected failure rate for top-quality aircraft parts and was expected to give a life of four months or better. For the SEO, the desired time between refurbishments is approximately nine months. For this case the failure rate was constrained not to exceed 2.2 times the applicable HI-REL failure rate (which is equivalent to certain standardized categories of military parts and components). For all longer life cases, normal space-quality parts (HI-REL) appear to be necessary in a practical system.

4.2.3.3 Minimum Subsystem Reliability. With the failure rate ratio fixed, the model for high cost subsystems, sometimes attempted to reduce reliability (and cost) by eliminating redundancy. Constraints were therefore imposed as needed

to prevent the reliability from going below that for a single thread (non-redundant) subsystem at that failure rate ratio.

4.2.3.4 Minimum Subsystem Cost. Insufficiencies in the integration of cost vs weight and cost vs reliability relationships into the model in some cases caused the analysis to call for impracticably low costs. This was prevented by placing floors under subsystem costs. This constrained the minimum subsystem cost to a level that was considered to be consistent with a value that might be achieved in the practical design case.

4.3 OPTIMIZATION ANALYSIS RESULTS

The cost optimization methodology described in Section 4.1 was exercised during the study through the use of the POP computer program. Thus it was possible to establish rapidly optimum programs for each given launch vehicle-payload combination in terms of minimum total program cost for a particular allowable payload weight and program duration. The major parameters to be considered in the cost optimization leading to the low-cost approaches were considered to be (1) total payload weight, (2) program time (duration), (3) launch cost, and (4) refurbishment/maintenance. For any change in either of these variables, the computer provides a new set of optimized program costs and the corresponding distribution of weight and reliability among the payload subsystems. The computer runs were planned so that the effect of each of these major parameters could be determined separately. A matrix showing the various computer runs and the parameters varied is shown on Fig. 4-10.

This section first describes the type and quantity of the computer output and then proceeds to evaluate the output data and establish results that have an important bearing on the development of low-cost payload approaches. These results led directly to the target costs and design goals summarized in Section 4.4.

PARAMETER VARIED	OAO			SEO		SRS			MO	
	ACE	LCE	SS	ACE/LCE	SS	ACE	LCE	SS	ACE	SS
Total payload weight:										
6000; 8000; 9500	3									
8000; 12000; 16000		3	3							
1000; 2100; 3000				3						
900; 1500; 2100					3					
400; 700; 1000						3	3	3		
1500;2500									(2)	2
Program time:										
1, 4, 6 yrs 8000 lb	+2									
1, 4, 6 yrs 12000 lb		+2	+2							
1, 4, 6 yrs 2100 lb				+2	+2					
1/2, 1, 2 yr 700 lb						+2	+2	+2		
1, 5 yr 2500 lb										+1
Maintenance/refurbishment:										
1 yr, 2 yr & no refurb. refurb. cost 0.3/0.5, maintenance, 0.1			+5		+5					
"Throwaway" -1, 4 & 6 yr.			+3		+3					
Launch Cost Apportionment:										
Zero, 1/2 & 1 x nominal		+2	+2				+2	+2		

Total Runs = 69

1 lb = .4536 kg

Fig. 4-10 Matrix of Parametric Analyses

4.3.1 Computer Output Format

One page of computer output summarizes the data for the optimum solution and compares it with the baseline payload. A typical sample of this page is attached as Fig. 4-11. A line-by-line explanation follows.

Line 1 - Data and page count

The date is helpful in correlating data changes and the introduction of practical design limitations with the computed results.

Lines 2 and 3 - Headings

The headings contain useful identifying information, usually parametric variations from run to run such as names of the payload and launch vehicle, total payload weight, mission and program durations, maintenance/refurbishment policy, launch vehicle cost, etc.

Lines 4 through 12 - Optimum Subsystem Data

These lines comprise a tabular summary of the following subsystem data at the optimum solution:

NUMBER	numerical order of the subsystems that compose the total payload
NAME	name of each subsystem
WEIGHT	subsystem weight, in pounds
REL	subsystem reliability, expressed as a decimal fraction
RCP	component reliability in each subsystem, expressed as a decimal fraction
RDT&E	RDT&E cost of the subsystem, in $1000
UNIT	unit cost " " " " " *
OPS	operations cost " " " *

---

* For the SEO payload, the unit and operations costs have been multiplied by four under the assumption that four flight programs will be in operation for this payload.

PAYLOAD EFFECTS ANALYSIS STUDY 30NOV70 DATA SCO/T30-CENT

FIX CR*S MSN DU 4YR NO MNT/REF CLV 822512 P2SE/B2/2+1K

NUMBER	NAME	WEIGHT	REL	RCP	RDTE	UNIT	OPS	REDUND R	REDUND U	FAIL RR
1	PL ASSY/INT	104.376	.9890	.9890	1878.391	730.667	3900.000	1.00000	1.00000	.0000
2	EXPERIMENTS	618.930	.8945	.8499	35073.517	6910.739	14028.000	1.18609	1.18609	.9693
3	STRUC/MECH	360.961	.9960	.9960	6283.969	4552.688	496.000	1.00000	1.00000	.0000
4	PROP/ATT CON	62.086	.9920	.9920	1833.319	692.000	256.000	1.00000	1.00000	.0000
5	GCN	241.029	.9856	.9610	6119.913	3396.928	1692.000	1.28801	1.28801	2.1901
6	TT+C	218.036	.9659	.9560	15731.629	5980.609	3940.000	1.07185	1.07185	2.2273
7	ELEC/PYRO	482.587	.9680	.9680	8396.195	5877.408	2452.000	1.00000	1.00000	.0000
8	ENVIR CONT	12.000	1.0000	1.0000	180.000	120.000	.000	1.00000	1.00000	.0000

CPROT	CPUNIT	CPUHR	CPOPS	CPROG	EXPMSL	RELP	WTP	ANUSE	ANLV
75491.870	28261.033	17856.191	26264.000	260906.459	2.67442	.80504	2100.000	1.58270	1.58270

BASELINE SYSTEM

NUMBER	NAME	WEIGHT	REL	RCP	RDTE	UNIT	OPS	REDUND R	REDUND U	FAIL RR
1	PL ASSY/INT	40.000	.9890	.9890	2887.000	868.000	3900.000	1.00000	1.00000	.0000
2	EXPERIMENTS	171.000	.8455	.8455	46571.000	8496.000	14028.000	1.00000	1.00000	1.0000
3	STRUC/MECH	138.000	.9960	.9960	7324.000	6444.000	496.000	1.00000	1.00000	.0000
4	PROP/ATT CON	51.000	.9920	.9920	1869.000	692.000	256.000	1.00000	1.00000	.0000
5	GCN	83.000	.9820	.9820	7803.000	4728.000	1692.000	1.00000	1.00000	.0000
6	TT+C	73.000	.9800	.9800	23531.008	9032.003	3940.000	1.00000	1.00000	1.0000
7	ELEC/PYRO	168.000	.9680	.9680	10356.000	6629.000	2452.000	1.00000	1.00000	1.0000
8	ENVIR CONT	12.000	1.0000	1.0000	180.000	120.000	.000	1.00000	1.00000	.0000

CPROT	CPUNIT	CPUHR	CPOPS	CPROG	EXPMSL	RELP	WTP	ANUSE	ANLV
100521.006	36804.002	21554.705	24264.000	306847.875	2.47922	.76965	726.000	1.70731	1.70731

Fig. 4-11 Example of Computer Printout

REDUND R REDUND U	ratio of the number of components in the optimum system to that in the baseline system as determined by the cost model used to compute the RDT&E and unit costs, respectively (both values are needed since practical design considerations may dictate the use of different models for these two cost components)
FAIL RR	failure rate ratio relative to that of the baseline subsystem (a value of zero is the result of choosing a cost model that ignores reliability tradeoffs for that particular subsystem)

Lines 13 and 14 - Optimum System Data

These two lines present the following optimum total system data:

CPRDT	RDT&E cost of the total program, in $1000
CPUNIT	sum of the subsystem unit costs, " "
CPUWR	unit cost of the entire payload with refurbishment and/or maintenance, apportioned over the original and all reactivated payloads, in $1000 (note the value quoted is irrelevant in this case since no refurbishment is involved).
CPOPS	sum of the subsystem operations costs, in $1000
CPROG	total program cost, in $1000
EXPMSL	expected useful life of each successfully launched payload, in time units appropriate to the particular payload (for SEO, the unit time is one year)
RELP	program reliability over the unit time period, expressed as a decimal fraction
WTP	total payload weight, in pounds
ANUSE	number of payloads required for the entire flight program; in the case of refurbishment and/or maintenance, the number of all reactivated payloads, plus the original one
ANLV	number of launch vehicles required for the entire flight program

Lines 16 through 26 - Baseline subsystem and system data similar to lines 4 through 14.

In addition to the optimized and baseline payload data described above, the sensitivities of the total program cost to small changes in the model coefficients are also printed out. These sensitivities are expressed as a ratio of

the percent change in cost to a percent change in the coefficient and as such they indicate to what extent uncertainties in the coefficients affect the total program costs.

4.3.2 Analysis Ground Rules

In the type of analysis considered here, where many programmatic variables are involved, it is important that some of these variables be either fixed or allowed to vary according to certain pre-determined rules. Thus, prior to starting the computer runs, it was agreed with the Fleet Analysis contractor (Aerospace) that the following ground rules and assumptions are realistic from a program point of view:

- All costs are in 1970 dollars.
- Standard shroud costs are included in the cost of the launch vehicle; if special shrouds are required (e.g., hammer-head), they are charged to the payload program.
- The weight and cost of deployment gear for the payload is charged to the launch vehicle.
- Launch vehicle costs are apportioned to the payload in the ratio of the payload weight to launch vehicle capability for that mission, assuming an 80% load factor, with the exception that the following launches are considered dedicated:
    - all alternate current
    - all OAO expendable
    - all SEO/Shuttle/Tug
- Total shuttle launch cost to the users is $3 million; for the Shuttle/Tug combination this charge is $3.7 million.
- Maintenance is on orbit and involves dedicated Shuttle/Tug launches for SEO; OAO maintenance launches are equal in cost to the initial payload launch (prorated).

- Refurbishment is on the Earth's surface and also involves dedicated launches for the SEO but not the OAO which are prorated.
- In the Shuttle/Tug mode, the Tug is taken to low Earth orbit by the Shuttle for each mission application.
- Operations cost reductions are estimated outside the computer but are included in the optimization of the total program costs; these reductions apply only to the Shuttle launched payloads, baseline operations costs are assumed for the expendable launch vehicle.

In the cases where the Shuttle was used as the launch vehicle, assumptions had to be made regarding the refurbishment cycle. Because the SRS was an inexpensive payload it was assumed that it would not be refurbished. For the OAO and SEO the nominal refurbishment schedule was once per year at a cost 0.3 times the unit cost. Parametric variations from this nominal schedule were investigated which included 2 year refurbishment, refurbishment at 0.5 times the unit cost and no scheduled refurbishment.

4.3.3 Optimization of Payload Weight

An essential tenet of the low-cost analysis methodology is that, contrary to conventional concepts, increasing the payload weight can result in cost savings. This is achieved by capitalizing on the reduced transportation costs (i.e., $ per lb to orbit) associated with the new classes of launch vehicles, through adopting a less sophisticated design approach, which usually results in a heavier payload. However, the growth in weight is limited by the capability of the launch vehicle being used. Further, as the "lowest cost regardless of weight" asymptote of the payload cost-weight-model is approached, only small payload cost-returns can be expected from large changes of weight. When the launch costs are prorated in proportion to payload weight these returns may be outweighed by increased launch costs. Thus, it becomes important to investigate the effect of total payload weight on the total program cost. This is done simply by changing the maximum weight constraint in the POP computer program.

In general, as a result of this exercise, a total payload weight could be selected beyond which no sensible cost reductions could be expected, and that weight then fixed while evaluating the effects of other program parameters. Some typical results are illustrated in Fig. 4-12 where total program cost is represented as a function of total payload weight. The results on the left are for the SRS payload used with each of the candidate launch vehicles and those on the right are for the OAO and SEO launched on the Shuttle.

For the total payload weights presented, minimum or near-minimum total program costs are shown. The apparent internal anomalies in the SRS case result from:

- fixed launch costs (dedicated launch) for Atlas/Burner II launch
- relatively high prorated costs, proportional to payload weight, for launch with SRM/Titan Core II/Agena launch
- relatively low prorated costs for Shuttle

As a result of these initial analyses, the following total payload weight constraints were placed on each payload for the ensuing computer runs:

SRS	700 lb (310 kg)
OAO	10,000 lb (4500 kg) for alternate current and low cost expendables 12,000 lb (5400 kg) for Shuttle
SEO	2,100 lb (950 kg)

In the case of the SEO, 2,100 lbs (950 kg) is approaching the maximum capability for the replacement (i.e., take payload to synchronous equatorial and return with the replaced payload) operational mode. For the OAO/alternate current and low cost expendable combination, the 10,000 lb (4500 kg) limit represents the limiting capability of the expendable launch vehicle, as provided by Aerospace.

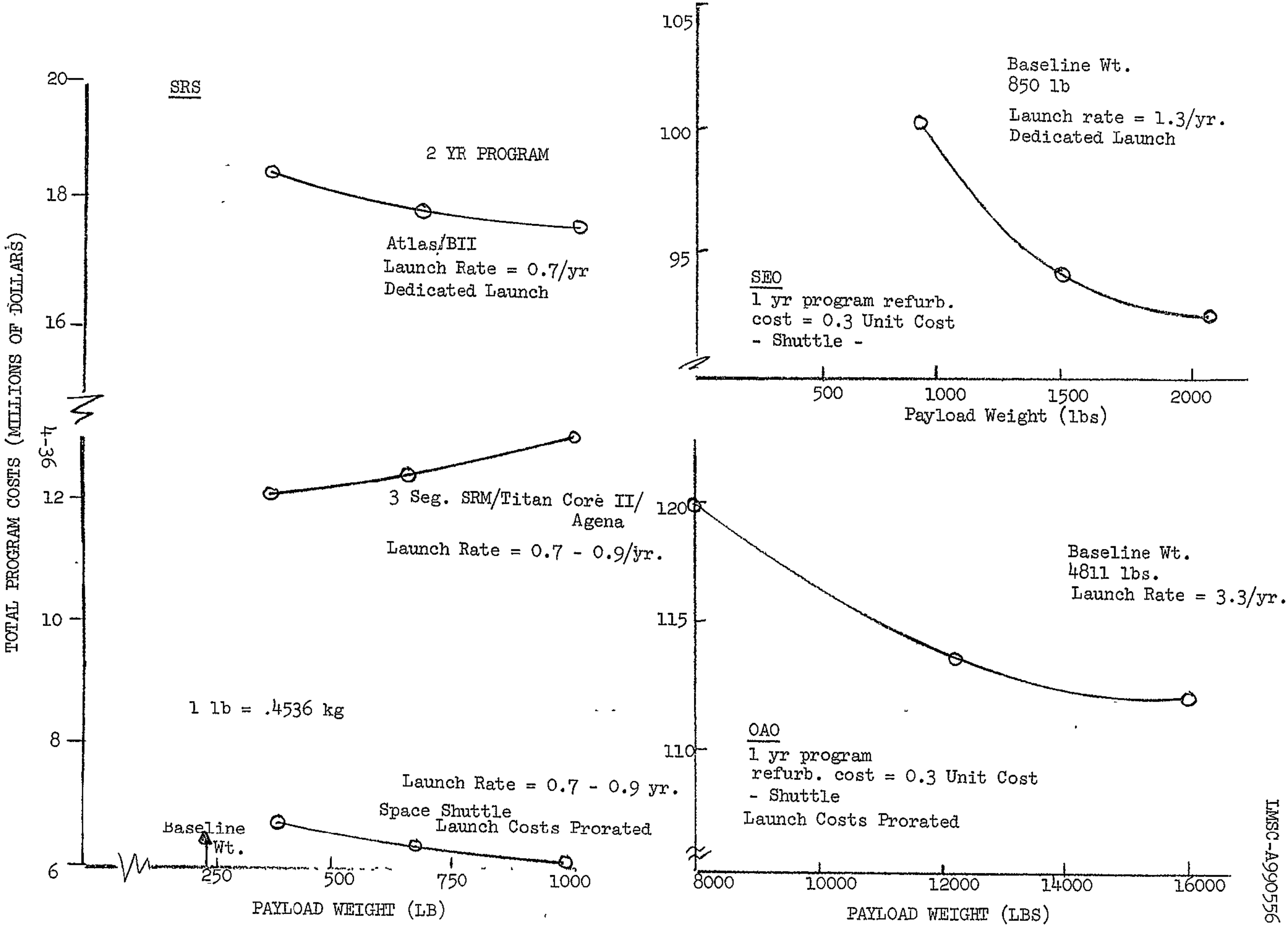


Fig. 4-12 VARIATION OF TOTAL PROGRAM COST WITH PAYLOAD WEIGHT

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4.3.4 Optimization of Expected Payload Life

The other important parameter which was selected on the basis of the parametric analyses was the expected payload life. This is influenced by

- the ratio of payload to transportation costs
- program duration and block time
- maintenance and refurbishment policies and costs

4.3.4.1 Transportation Costs. When the payload cost is low relative to the transportation costs, as in the case with SRS, it becomes potentially cost effective to invest in increased payload life and thereby reduce the required number of launches. This approach is subject to explicit cost tradeoff analysis, and the resulting optimum payload life will vary with costs per launch. When the payload cost is high relative to transportation costs, as is the case with OAO, transportation costs exert little effect on optimum payload life and the cost optimization will be driven by a requirement to minimize payload costs for the total program.

4.3.4.2 Program Duration and Block Time. The mission program may in some cases be divided into a series of "blocks". At the end of each block the experiment package may be changed and the payload, in the case of Shuttle transportation, refurbished. These block times may be dictated by the nature of the experiment or by practical considerations of achievable payload life. In the case of OAO and SEO the nominal block time, or refurbishment frequency, was once per year.

With expendable launch vehicles, because there was no refurbishment, optimum payload life was directly affected by program duration. For the OAO with expendable launch the parametric analyses suggested that a flight-duration of approximately 1 year was optimum for any program time in excess of two years. For the SEO, the analyses tended to suggest a payload life of two years as optimum; design considerations suggest, however, that it may not in fact be practical to extend payload life appreciably beyond 2 years for this type of

payload. The SRS, because it is inexpensive, is very sensitive to program duration, with the result that a payload life of about 15 months appears desirable even with a program duration of only two years.

With Shuttle launch and scheduled refurbishment (as with OAO or SEO with Shuttle), the optimum payload life is bounded by the refurbishment schedule. With the nominal annual refurbishment, the payload life indicated by the parametric analysis was approximately 4 months for OAO and 0.8 yrs for SEO. This is further discussed below.

4.3.4.3 Maintenance and Refurbishment Policies. With Shuttle launch and the provision of maintenance and refurbishment capabilities, the optimum payload life becomes strongly dependent on the refurbishment schedule. The results from the parametric analyses may be summarized as follows, for a program time of four years or more.

Payload	Refurbishment Schedule	Payload Life
OAO	1 yr	4 months
	2 yr	8 months
	None	1 yr
SEO	1 yr	0.8 yr
	2 yr	1.4 yr
	None	2 yr

4.4 COST TARGETS, RELIABILITY GOALS, WEIGHT ESTIMATES

In addition to providing some insight into the cost consequences of low-cost design techniques, a major output of the parametric analyses was the formulation of preliminary cost estimates (cost targets) and weights of low-cost payloads and design goals for the ensuing design effort. Because of the variation of the non-recurring and recurring costs with such program factors as program time, apportioned launch costs, and maintenance or refurbishment

frequency, no single version of the low-cost payload exists that can be applied across the mission spectrum in a completely optimum manner. Therefore, in choosing those design points for which the target costs could be quoted, it was important to consider that the low-cost or target payload is (1) consistent with the baseline mission requirements, (2) relatable to the baseline hardware, and (3) compatible with the baseline flight duration. In addition, the operating life must be compatible with the failure rate characteristics of available hardware. Thus, for the expendable payloads the low-cost targets were defined for low-cost payloads with the same flight-durations as the baseline nominal values. With the Shuttle, however, it is important to guide the payload design towards the optimum maintenance mode; for the low-cost OAO payloads, target costs were established for systems consistent with flight durations of 4 months and 1 year.

In the case of SEO, a flight-duration of 0.8 yr was selected (close enough to the 1-yr baseline to cause no design problems). These flight durations for the reusable payloads are compatible with the results of the maintenance/refurbishment analyses reported in Section 4.3.6. Additional data were also provided for a 2-yr SEO as this duration was a better match to many missions in the total program.

In addition to the baseline-equivalent 6-month SRS, target costs were established for the SRS-Shuttle low-cost payload associated with a flight duration of 15 months, since this version has particular application to the NASA mission model.

The optimized target data for the Mars Orbiter are relevant to a single shot mission, consistent with the baseline mission duration, and are provided for the Space Shuttle only.

The above data were provided to Aerospace as interim working data, subject to verification by the subsequent design analyses, to enable them to make an early start on their Fleet Analyses. A sample of these data, for the 1-year OAO on the Space Shuttle, is shown in Fig. 4-13. A payload level summary of the data is shown in Fig. 4-14.

PAYLOAD - OAO-B LAUNCH VEHICLE - SPACE SHUTTLE FLIGHT DURATION 1 YEAR

Subsystem	COMPUTER OUTPUT			CONSERVATIVE ESTIMATE* ± TOLERANCE				
	RDT&E Cost	Unit Cost	1 Year Operations	RDT&E Cost	RDT&E Est. Tolerance	Unit Cost	Unit Est. Tolerance	Unit Opn's. ±30%
Payload Adapter	0	0	0	0	-	0	0	0
Experiments	4800	5580	1263	6000	± 20%	6200	± 10%	1804
Structures & Mechanisms	5865	3780	9	6900	± 15%	4200	± 10%	13
Electrical & Pyro	10400	2340	315	13000	± 20%	2600	± 10%	450
Stabilization & Control	40000	8100	2247	50000	± 20%	9000	± 10%	3210
Propulsion & Attitude Control	1737	228	120	1930	± 10%	240	± 5%	171
Communication, Data Processing & Instrumentation	24000	3600	1611	30000	± 20%	4000	± 10%	2301
Environmental Cont.	3080	1140	315	3850	± 20%	1200	± 5%	450
Non-Allocated Costs	800	1125	900	1000	± 20%	1500	± 25%	1286
Payload Total	90682	25893	6780	112680	± 19%	28940	± 10%	9685

* Recommended for costing purposes.

Fig. 4-13 Example of Subsystem Cost Estimates for Low-Cost Payloads (1970 $ Thousands)

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PAYLOAD/LAUNCH VEH.	FLIGHT DURATION	(LB) WEIGHT ESTIMATES	COMPUTER ESTIMATES ($1000) RDT&E	UNIT	OPS**	CONSERVATIVE LOW-COST ESTIMATES ($1000) RDT&E	FIRST UNIT	OPS**
SRS/SLV3C/BII	6 mo.	540 + 43*	7691	966	190	7806 ± 8%	1001 ± 11%	
SRS/3SEGSRM/AGENA	6 mo.	540 + 43*	6044	824	190	7646 ± 7%	984 ± 11%	
SRS/SS	6 mo.	540	3502	670	150	4519 ± 23%	821 ± 19%	187 ± 20%
SRS/SS	15 mo.	540	3783	756	275	4876 ± 22%	931 ± 20%	343 ± 20%
OAO(B)/SLV3C/CENT.	1 yr.	5880+300*	159638	34562	11220	146810 ± 11%	33524 ± 7%	
OAO(B)/TIII/L2	1 yr.	7330+300*	134877	28699	11220	131660 ± 14%	32550 ± 9%	
OAO(B)/SS	4 mo.	10330	70967	22905	2260	88860 ± 20%	26268 ±14%	3230 ± 30%
OAO(B)/SS	1 yr.	8300	90682	25893	6780	112680 ± 19%	28940 ±10%	9685 ± 30%
SEO/IIID CENT.	1 yr.	1515+114	79155	7576	7993	95332 ± 19%	8100 ± 9%	
SEO/SS-TUG	0.8 yr.	1595	44548	5983	2530	57130 ± 22%	7266 ±18%	3615 ± 30%
SEO/SS-TUG	2 yr.	---	---	---	---	68640 ± 22%	10810 ±18%	6530 ± 30%
MARS ORBITER/SS-TUG	0.9 yr.	1970	52633	9639	12985	68545 ± 23%	11010 ±13%	18551± 30%

* LAUNCH VEHICLE INTEGRATION HARDWARE

** OPS COSTS FOR APPROPRIATE FLIGHT DURATION

1 lb = 0.4536 kg

Fig. 4-14 SUMMARY OF INTERIM COST AND WEIGHT DATA

4.5 UPDATE OF COMPUTER OPTIMIZATION METHODOLOGY

The cost optimization methodology described in Sections 4.1 and 4.2 proved extremely effective as a tool for investigating payload effects on launch vehicle economics and enabled rapid investigation and clear illumination of the economic impact of the payload on the total program cost. The mathematical model used and the associated computer program were, however, developed and checked out in a rather short time period. Because of the time constraint, only immediately necessary options were implemented in the initial model.

The mathematical model was designed to be highly modular, and capable of being readily modified or extended. Following completion of the low-cost payload designs and their costing, another look was taken at the model with the intent of adding some extensions in the areas of (1) repair and refurbishment modelling, and (2) reliability modelling. These extensions have been defined (as described following) but have not yet been programmed for the computer.

4.5.1 Repair, Refurbishment and Standby System Options

As the analyses progressed it became evident that extended flexibility was desirable in the model in these areas. For example, in modelling refurbishment and repair of OAO it was assumed that two roundtrips would be required for refurbishment, but one only for repair. However, analyses of the refurbishment operation subsequently showed that refurbishment on orbit, which required only one roundtrip, was feasible and preferable. Also, some missions require continuity of service, leading to a probable requirement for payloads on orbital standby. The model has therefore been extended to provide the following capabilities:

- clear separation of the costs of placement, retrieval and repair on orbit, allowing for use of different modes and associated transportation for these functions
- provisions for "as required" as well as scheduled refurbishment

- provision for refurbishment instead of repair if a failure occurs at a time "close" to the scheduled refurbishment time

- provision for orbital standby systems

The necessary extensions in the model to provide these capabilities are at the system level and consist primarily of providing a more detailed breakout of cost elements.

4.5.1.1 System Model. The total program cost C_{PR} is divided into the following elements:

payload RDT&E costs		C_r
total payload new unit costs		$C_{ut/n}$
total refurbishment costs		$C_{ut/ref}$
total maintenance/repair costs		$C_{ut/m}$
total payload operations costs		$C_{ops/t}$
total costs of standby orbital system(s)		$C_{sb/t}$
total transportation costs	- placement	$C_{LVt/h}$
	- retrieval	$C_{LVt/r}$
	- maintenance/ repair	$C_{LVt/m}$

Hence, we have C_{PR} equal to the sum of all of the above elements. Of these, the RDT&E costs C_r are derived directly from the payload cost model. The remainder are discussed in turn below.

4.5.1.2 New Unit Costs - for total new unit costs:

$$C_{ut/n} \quad = \quad C_{u/n} \; N_{nu/P_L}$$

where $C_{u/n}$ = cost of single new unit

P_L = probability of payload surviving launch in functioning or repairable condition, assumed unity for Shuttle

N_{nu} = initial number of active payloads required in system plus any scheduled replacements required after some maximum number of refurbishments

Note that with an expendable launch system N_{nu} will cover all replacements required.

4.5.1.3 Refurbishment Costs.

$$C_{ut/ref} = C_{u/n} R_{rf} N_{rf}$$

where R_{rf} = ratio of single refurbishment cost to single new unit cost

N_{rf} = number of refurbishments required

If refurbishment is on a scheduled basis, then

$$N_{rf} = N_{S/a} (T_{PR}/T_{RF} - 1)$$

where $N_{S/a}$ = number of active satellites required in system

T_{PR} = duration of program

T_{RF} = scheduled time interval between refurbishments

If, on the other hand, refurbishment is only effected as required,

$$N_{rf} = N_{S/A} \left\{ T_{PR}/E\,(T_{RF}) - 1 \right\}$$

where $E(T_{RF})$ = expected time between refurbishments

4.5.1.4 Maintenance/Repair Costs.

$$C_{ut/m} = C_{u/n} R_m N_m$$

where R_m = ratio of repair cost to single unit cost of new payload

N_m = expected number of maintenance operations required

As a matter of practical operations, one would expect that if a failure occurred shortly for a scheduled refurbishment the failure would be corrected refurbishment rather than by "repair". If this option is exercised within a time $\Delta_m t$ of the scheduled refurbishment time then

$$N_m = N_{S/a} (T_{PR}/T_{RF})(T_{RF}/E(T_m) -1) - N_{rf} \Delta_m t$$

where $E(T_m)$= expected time between failures of an active payload

4.5.1.5 Operations Costs. The payload-peculiar operations costs, over and above standard transportation costs included in the transportation price, may contain elements essentially proportional to program time and also to the number of placements. Thus one may in general approximate those as

$$C_{ops/t} = AT_{PR} + B (N_{rf} + N_{nu})$$

4.5.1.6 Non-Transportation Costs of Standby Orbital Systems. It will presently be assumed that

- a failed active payload is replaced by a standby already in orbit, which is in turn replaced by a newly launched standby
- a payload which fails while on standby is repaired

Thus, the incremental cost of the standby system, over and above costs accounted for already in the active system, is

$$C_{sb/t} = C_{u/n} (N_{S/S} + N_{m/S} R_m)$$

where $N_{S/S}$ = number of standby satellites required in system

$N_{m/s}$ = number of satellites which fail while on standby

$= N_{SS}\, T_{PR}\, \lambda_{sb}$

where λ_{sb} = failure rate per payload while on standby

4.5.1.7 Transportation Costs. The transportation costs for placement, retrieval and maintenance are simply the products of the respective costs for the single operations and the respective numbers of operations required

$$\text{placement} - C_{LV/p} = C_{LVu/p}\,(N_{S/a} + N_{S/S} + N_{rf})/P_S$$

$$C_{LV/r} = C_{LVu/r}\, N_{rf}$$

$$C_{LV/m} = C_{LVu/m}\,(N_m + N_{m/S})$$

where the subscript "u" denotes a single operation and

P_S = probability that satellite will be functioning when placed

4.5.2 Payload Reliability, Expected Life and Flight Value

4.5.2.1 Basic Tradeoffs. In the model described in Section 4.1, the "expected cost" of performing a given mission or a given sequence of mission blocks was minimized. The mission was considered as a requirement to be on orbit, performing some function, for the prescribed time. The value of each flight was described by the expected time over which the payload would operate on that flight before failure and cost minimization was effected by trading (a) the increased payload non-recurring and single unit costs associated with extending payload life against, (b) increases in transportation costs, and (c) numbers of units required with shorter payload life.

In general, when the function is performed (and any data recovered) on a continuing basis:

$$\text{expected value per flight} - M_{SL} = \int_0^T R(t)\, V(t)\, dt$$

where $R(t)$ = probability of survival to time t

$V(t)$ = value of data gathered at time t

T = maximum time of interest

$V(t)$ will presently be assumed equal to unity. T will usually be either the total mission time or, if refurbishment or change of experiment is required, the scheduled time between refurbishments.

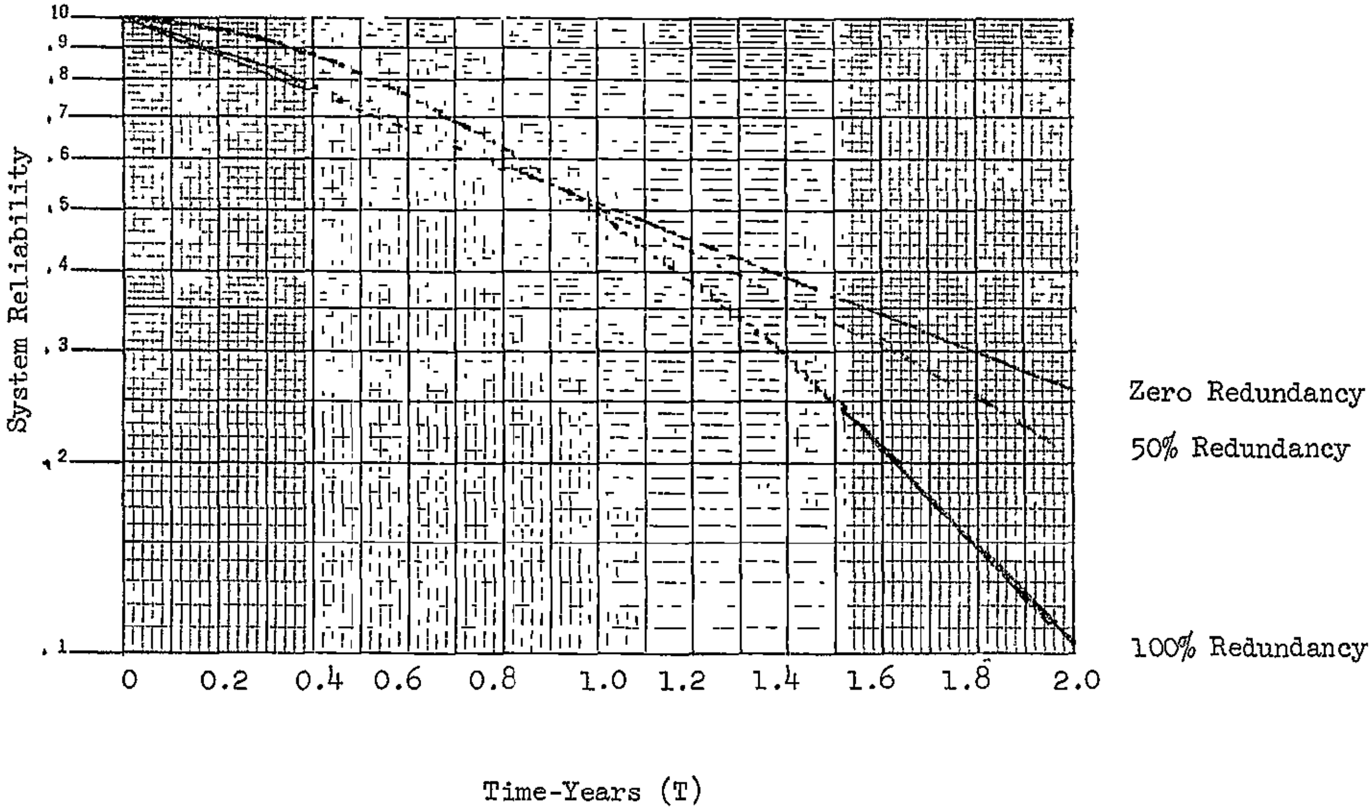
4.5.2.2 Effect of Redundancy. In Section 4.1 an exponential form was assumed for reliability $R(T_R)$ as a function of scheduled flight duration T_R. This permitted use of a very convenient explicit form for the expected value per flight, as follows:

$$M_{SL} = \left(1 - R^{\frac{T_M}{T_R}}\right) / \ln R\,(T_R)$$

where T_M = maximum permissible duration, at which the reliability-life curve would be truncated

T_R = scheduled flight duration for estimate of R

With appropriate truncation to allow for such phenomena as wearout and parametric drift the exponential assumption is legitimate for present purposes for a moderately redundant system (e.g., 50 percent of components replicated) but becomes questionable at redundancy fractions approaching 100 percent. This is illustrated in Fig. 4-15, in which three systems are compared, each of which has a reliability of approximately .50 at one year. The 100 percent redundant system has a component failure rate 4.5 times that of the single thread system, but achieves equal reliability at one year by the extensive redundancy. By this time, however, three of the redundant components will, on the average have failed. This leaves the system seriously degraded, with only seven of the ten components still effectively replicated and with a correspondingly poor life expectancy.



- Random failure only - no wearout
- System with 10 functional elements
- 100% Redundant System has active replicate of each element
  50% Redundant System has active replicate of 5 elements
- Element failure rates - 0.067 per year for system with zero redundancy
  - 0.13 per year for system with 50% redundancy
  - 0.30 per year for system with 100% redundancy

Fig. 4-15 Example of Effect of Redundancy on Reliability vs Scheduled Life

A brief review of this effect led to the following conclusions:

- with redundancy up to 100 percent the expected assumption appears satisfactory (M_{SL} within 5 percent) if $T_M < 2\ T_R$ - i.e., with truncation no later than twice the nominal flight duration

- with a redundancy of 50 percent the exponential assumption appears satisfactory as long as $T_M < 4\ T_R$.

If, therefore, the first cut optimization leads to a system lying outside these constraints, consideration should be given to introducing a system-particular expression for the payload expected life M_{SL} as a function of failure rate and redundancy. It will usually be feasible to do this for a particular payload, even including such effects as duty cycle and dormant or quiescent failure rates, and provide the data to the computer in either tabular or algebraic form.

4.5.2.3 Subsystem Reliability Mode. The effects of redundancy on subsystem reliability was modelled during the study at the subsystem level -- that is, redundancy was assumed to consist of providing a complete replicate subsystem. This was used in order to be conservative. It is, however, unrealistic. The alternate model, corresponding to redundancy at component level, is therefore preferred. For unit costs this was

$$\frac{\text{cost}}{\text{baseline cost}} = \frac{\text{weight}}{\text{baseline weight}} = \frac{\ln N_j - \ln(-\ln R_j)}{\ln N_j - \ln(-\ln R_{jb})}$$

where N_j = number of functional modules in subsystem.

This is an approximate form of the more precise relation

$$\text{cost}\ \alpha\ \text{weight}\ \alpha\ \ln\left[1 - \left\{R_j(m)\right\}^{1/N_j}\right]$$

where m = # modules in redundant system/# modules in single thread system

For RDT&E costs the corresponding expression is

$$\frac{\text{cost}}{\text{baseline cost}} = 1 - \alpha_3 + \alpha_3 \quad \frac{\ln N_j - \ln(-\ln R_j)}{\ln N_j - \ln(-\ln R_{jb})}$$

Section 5
DESIGN OF LOW-COST PAYLOADS

5.1 LOW-COST PAYLOAD DESIGN CRITERIA

5.1.1 General Low-Cost Design Philosophy

A new design approach, not constrained by traditional weight or volume limits, nor by the one-chance only hazards of expendable launches; has been developed for future payloads to be launched and supported by the Space Shuttle. Application of these approaches to the unmanned payloads planned for the 1978-1990 era will result in significant cost savings relevant to "baseline" programs wherein expendable launch vehicles and payloads are utilized. These payloads will be "low-cost" because of the "Payload Effects" that are the subject of this report. These effects are discussed in following paragraphs.

5.1.1.1 Shuttle Payload Capability. The payload capability of the Space Shuttle for near-earth missions will be very large relative to the probable weight of the spacecraft it will transport to orbit.

Therefore, in general, the design of spacecraft for Shuttle launch will not be constrained by weight limitations.

Similarly, the design of Shuttle-launched spacecraft will not be constrained by volume limitations in the large 15 ft (4.6m) diameter, 60 ft (18.3m) long cargo bay of the Space Shuttle.

The spacecraft designer will not normally be concerned with the total payload volume of the Space Shuttle cargo bay. He will, rather, pay careful attention to the specific volume required to obtain favorable low cost in the spacecraft to be designed. The elimination of volume constraints will be a significant factor in the reduction of spacecraft costs.

Also, the design of spacecraft for Shuttle launch will be strongly influenced by the requirement to provide for access to the spacecraft equipment modules to accomplish their removal and replacement within the Shuttle cargo bay. Exchange of modules when the spacecraft is elevated out of the cargo bay by deployment mechanisms will be possible, but designs that permit the exchange within the cargo bay will be preferred since they simplify the Shuttle space crew activity and enhance personnel safety.

5.1.1.2 Shuttle Environment for Payloads. The loads and the shock, vibration and acoustic environments to which payloads will be subjected in the cargo bay of the Space Shuttle will be, in general, less severe than those associated with expendable launch vehicles. In addition, not being constrained by weight and volume limitations, Shuttle-launched spacecraft structures and equipment will be designed more rugged and less sensitive to these loads and environments. Consequently, the need for intensive analysis and tests to assure the survival of spacecraft through launch will be reduced and hardware development costs will be reduced.

5.1.1.3 Payload Checkout in Orbiting Shuttle. In the design of payloads for launch by the Space Shuttle there will be a number of departures from past practice that will lead to the reduction of testing costs. The firm knowledge that no payload will be committed to its mission orbit without a successful system checkout in the cargo bay of the orbiting Shuttle, and the knowledge that a malfunctioning payload can be recovered for repair in orbit or for return to earth will minimize the historical fear of the consequences of failure that has led to (1) overdesign, (2) over-specification, and (3) incorporation of redundant equipment and backup systems not clearly required for mission success.

Additionally, the "increased risk" approach available with a Shuttle-launched payload will permit relaxation of the testing criteria established for expendable-launched payloads.

5.1.1.4 Payload Modularity. An additional benefit to payload design deriving from increased payload capability (Shuttle and new expendables) will be true

modularity of equipment packaging, which has been inhibited by the weight and volume constraints of historical expendable launch vehicles. Well-designed equipment modules of a payload system may be removed from the system and replaced by an equivalent module without perturbation of the system function or calibration. Such interchangeability of modules will simplify and reduce the costs of system level testing. The following guidelines were made applicable to the design of equipment modules for the low-cost payloads.

- Divide payload subsystems into minimum quantity of modules consistent with:
  - Maximum weight/size which can be readily installed or removed by a Shuttle crewman
  - Maximum cost of a single module not to exceed 10 percent of payload recurring cost.
- Segregate components which have high probability of replacement from those which have higher predicted life.
- Establish operating tolerances on individual modules so that module replacement will not require payload recalibration.
- Provide simple functional and mechanical interfaces between modules.
- Provide for easy access to and removal/installation of modules without need for special tools.

Modularization of equipment installations will also contribute to cost savings in spacecraft ground operations prior to launch.

5.1.1.5 Payload Design vs Program Cost. Payload designers of future payloads must be made aware that payload design at all levels; system, subsystem, component, or part; significantly affects all categories of program costs. Payload designers must consider the cost impact of all design decisions if program

costs are to be minimized. This will require a greater alertness among designers of the cost consequences of their decisions; this will occur only through training and improved communication among all organizations contributing to the execution of payload hardware programs.

5.1.1.6 Engineering Costs. Engineering costs are directly related to the complexity of the designs to be created. To reduce engineering costs, payload designers must strive for design simplification, beginning in the concept design phase and continuing through the detail design phase. Much of the complexity of spacecraft has been due to overspecification of functional requirements and to limitations on the size, weight, and power consumption of equipment. When the Space Shuttle replaces expendable launch vehicles for the transportation of space payloads, the factors which have led to overspecification will lose much of their significance, specifications will be less restrictive, and payload designers will have the freedom to create simpler, less costly designs.

5.1.1.7 Low-Cost Materials. In general, for comparable applications, materials that are inexpensive to buy and fabricate, such as steel and aluminum, are heavy; and lighter materials, for example titanium and beryllium, are expensive. In the past, payload weight limitations have prevented the free use of heavier materials and simple methods of fabrication; and the costs of spacecraft have been higher than they would have been without weight limitations. In designing future low-cost spacecraft, full advantage should be taken of the increased payload capability of the Shuttle or high-capability new expendables in use of low-cost materials and fabrication methods.

5.1.1.8 Pre-Qualified Equipment. Most space programs have made some attempt to use pre-qualified equipment to obtain the obvious cost savings. However, existing qualified equipment has often been considered and rejected because it was not optimized functionally, or in size and weight, for the contemplated application.

The relaxation of weight, volume, and power constraints on spacecraft design will permit designers to consider a much wider range of qualified or "off-shelf"

equipment for a given application, and to select equipment that would have been rejected as overdesigned, oversize, or overweight under past standards. Much more emphasis must be placed on avoiding new RDT&E costs in future space programs.

5.1.1.9 Standardized Equipment. The development and qualification of standardized equipment for spacecraft promises major overall savings for the Nation's space programs. There has been very little standardization of equipment in the past because the emphasis has been on the optimization of equipment function, size, and weight for the requirements of each space program. When such optimization is no longer required because of the elimination of constraints on payload weight and volume, the development of standardized equipment can proceed with reasonable assurance that it will be more widely used throughout the national space program. The Government should consider establishment of standard design requirements for commonly used types of spacecraft equipment, perhaps establish specific industry sources for such equipment, and specify that the selection of equipment for new spacecraft shall, if technically possible, be from among the available standard equipment.

5.1.1.10 Low-Density Packaging of Electronic Equipment. Electronic equipment for space programs has evolved in the direction of reduced size and weight and increased packaging density. In general, this evolution has been accompanied by cost escalation (with the exception of substitution of solid-state for vacuum-tube elements). The relaxation of weight and volume constraints in the next era should permit the development of simpler, lower-density packaging techniques for electronic equipment; resulting in the reduction of RDT&E and recurring costs of such equipment.

Low-density electronic packaging saves labor costs in design, modification, fabrication and assembly, repair, and inspection.

5.1.1.11 Tooling. The costs of tooling for space payloads are directly related to the tolerances specified for the assembly and alignment of structures

and for the installation of equipment. The larger, heavier structures of the new low-cost payloads should not require as much tooling as the lighter, more flexible structures of historical payloads.

Also, tooling costs can be reduced if, in the design of a payload structure, a single structurally stable plane can be established on which all equipment modules requiring alignment can be mounted.

5.1.1.12 Manufacturing. Manufacturing costs can be reduced if payload designers avoid sandwich materials, chemical milling, machine contouring, and other techniques for weight control commonly used in the design of historical spacecraft but unnecessary in the design of the new payloads.

Low-density packaging of equipment into modules and of modules into payload structures will help to reduce manufacturing costs by providing easy access to all components and reduced assembly costs.

The manufacturing costs of electronic equipment can be reduced by low-density packaging because of improved access for the placement of parts and for inspection, which in turn results in the reduction of scrap and rework.

5.1.2 System Design of Low-Cost Payloads

5.1.2.1 System Performance and Design Requirements. Over-specification of system requirements has been a significant factor in escalating the costs of space programs, and it begins with the System Performance and Design Requirements Specification prepared by the government program office and/or the contractor. Cost consciousness should be fostered among the scientists, engineers and managers responsible for program planning, with the objective of obtaining requirements specifications based on realistic cost/value analysis.

5.1.2.2 Simplified Documentation. Much of the escalation of the costs of space programs is attributable to the customer and contractor documentation requirements.

Program planners should carefully evaluate documentation needs, and impose only those requirements essential to the orderly execution of the development/test program.

When contractual documentation requirements permit, simplification of contractor engineering documentation can result in significant savings.

5.1.2.3 Simplified Equipment Specification. It is not just space program planners who may over-specify requirements. Engineering organizations also write design specifications that are more restrictive than they need to be; because it is safer (but more costly) to err in that direction. Government program offices could encourage contractors to specify lower equipment performance and other design requirements, and contractors, in turn, could establish effective review procedures to control over-specification of requirements for both in-house manufactured and purchased or subcontracted equipment. When spacecraft are being designed for launch by the Space Shuttle, the pressures that lead to overspecification should be less severe because the Shuttle will provide for in-orbit checkout and revisit and thereby allow use of a payload with higher initial risk.

5.1.2.4 Reduced Testing Requirements. Typically, spacecraft equipment and assembled spacecraft have been subjected to very comprehensive, rigorous, and often repetitive testing programs to establish the level of confidence felt to be necessary for the making of launch decisions. Such programs have been costly to execute and often costly in worn-out or damaged equipment. The fear of spacecraft failure during launch or early in orbital flight and of the consequences of such a catastrophe has been the principal motivator for traditional testing programs. Much of this apprehension can be alleviated by the Space Shuttle which makes possible in-orbit checkout of spacecraft before they are committed to orbit, monitoring of their early orbital performance, and recovery of the payload for repair in orbit or return to earth in the event of early malfunction. Once the consequences of early hardware failures have been lessened, it should be possible to reduce significantly the scope and cost of testing programs.

In addition, load testing of the structures and mechanisms of low-cost spacecraft may be significantly reduced since they may be designed without weight constraints and with high factors of safety.

5.1.2.5 System Design Guidelines. The following are a few guidelines for payload system designers, which have been implemented in the design of low-cost payload systems during the performance of the Payload Effects Analysis study and which have contributed to the cost savings estimated to accrue from those designs:

- Do not overspecify performance requirements.
- Select a simple spacecraft configuration, taking full advantage of the payload weight and volume capability of the Space Shuttle.
- Select the simplest systems that will meet specification requirements to reduce design, analysis, fabrication, and testing efforts.
- Establish reliability goals based on the in-orbit checkout capability of the Space Shuttle.
- Limit equipment redundancies and backup operating modes to those actually required by reliability goals.
- Avoid new technology developments but exploit new technology that has been reduced to hardware.
- Minimize command and data requirements.

5.1.3 Design of Low-Cost Payload Subsystems

5.1.3.1 Structures and Mechanisms Subsystems. When weight and volume constraints are relaxed or eliminated, significant reductions in the cost of spacecraft structures and mechanisms can be made by using the following guidelines in design.

a. Select a simple spacecraft configuration that requires only a simple structure.

b. Provide volume for low-density equipment installations to simplify installation design and to insure complete accessibility of equipment.

c. Use high factors of safety (three or greater) for sizing structural elements to reduce design and analysis efforts and to reduce or eliminate static load testing.

d. Increase dimensional tolerances to simplify tooling, and to reduce fabrication, assembly and quality control efforts.

e. Simplify structural elements and minimize the use of machined parts to reduce design, analysis, fabrication and assembly efforts.

f. Use commercially available grades and sizes of aluminum sheet and extrusions for most structural elements.

g. Minimize the use of sandwich materials.

h. Do not use beryllium, composite materials or other high-cost materials.

i. Eliminate deployment mechanisms whenever the launch vehicle payload envelope permits fixed installation of solar panels, antennas, sensors and other equipment.

j. Avoid sophistication and miniaturization of mechanisms.

5.1.3.2 Experiment Subsystems. Experiment equipment for space missions, like spacecraft structures has been constrained by the limited payload weight and volume capability of launch vehicles. When weight and volume constraints are relaxed, the costs of Experiment Subsystems can be reduced by using the following guidelines in design.

a. Select simple experiment package configurations, taking full advantage of the greater payload capability.

b. When experiment thermal control requirements differ significantly from other spacecraft subsystem requirements, isolate the experiment to simplify thermal control of both the experiment and the spacecraft.

c. Design for in-orbit maintenance of experiment installations by modularization of equipment.

d. Design low-density experiment installations with provisions for additions of or changes to experiment equipment.

e. Avoid mechanisms that are not self supporting in 1-g.

f. Design low density electronic packages to reduce design, development and manufacturing costs.

g. Eliminate in-flight adjustments.

h. Avoid use of high-cost material such as beryllium for weight reduction.

i. Avoid miniaturization for weight and volume reduction.

5.1.3.3 Stabilization and Control Subsystems. Stabilization and Control Subsystems are usually the most costly subsystems of spacecraft because of the complexity of the sensors, actuators, and electronic equipment required for their mechanization. The design of such subsystems have been constrained by weight, volume, and power limitations, and have been complicated by redundancy and backup modes incorporated to maximize probability of mission success. Efforts to reduce the costs of Stabilization and Control Subsystems should seek to reduce their complexity and to eliminate the cost penalties that derive from weight control requirements. Some guidelines for the design of Stabilization and Control Subsystems are as follows:

a. Select the simplest system, that will meet specification requirements, to reduce design, analysis, fabrication, and testing efforts.

b. Do not overspecify component performance requirements.

c. Limit equipment redundancies and backup operating modes to those specifically required by reliability goals.

d. Simplify equipment design by taking full advantage of the greater weight and volume capability afforded.

e. Avoid new technology development and exploit technology that has been reduced to hardware.

f. Tradeoff the use of a general-purpose computer for Stabilization and Control and Data Processing functions against alternate mechanizations.

g. Increase the volume of electronic equipment (x2 or more) to reduce packaging density, thus reducing design, manufacturing, and inspection costs.

h. Reduce stress on parts and/or use larger higher-rated parts, thereby in circuit design, increasing confidence in performance, and reducing testing costs.

i. Minimize command and data requirements.

j. Design for in-orbit maintenance by modularization.

5.1.3.4 Communication, Data Processing, and Instrumentation (CDPI) Subsystems. The Communications, Data Processing, and Instrumentation subsystem includes much complex electronics equipment, and to accomplish cost savings in the design, development, and procurement of CDPI subsystems means must be found to reduce the quantity of equipment required and also to reduce its complexity. Digital computers can perform programming, control and data processing functions, and the cost saving potential of computers should be thoroughly evaluated

in the system design of a new spacecraft. Relaxation of weight and volume constraints here again permits the design of low density electronic equipment that is inherently less costly to design, develop, and produce than high density electronics.

Standardization of CDPI equipment will relieve space programs of the costs of design and development and will result in significant savings.

The design guidelines for Stabilization and Control subsystems are equally applicable to CDPI subsystems.

5.1.3.5 Electrical Power Subsystems. Most Electrical Power subsystems of spacecraft employ solar arrays for the conversion of solar energy to electrical energy and batteries for the storage of electrical energy. They include equipment to control the charging of the batteries, the conversion of voltages, and the distribution of electrical power.

Changes in the design of spacecraft batteries do not promise significant cost savings. However, if battery size variations can be limited and standardized, cost savings will accrue to the national space program.

The cost of equipment for regulation and control of electrical power can be reduced by simpler, low density packaging and by standardization.

Relaxation of weight and volume constraints will result in lower costs for solar arrays, which can be of simpler design and can be made of low-cost structural materials and lower-rated, less costly solar cells.

The design guidelines for Stabilization and Control Subsystems are equally applicable to the design of Electrical Power Subsystems.

5.1.3.6 Attitude Control Subsystems. In this document, by definition, the Attitude Control subsystem is limited to systems that produce torques by the

expulsion of mass for orientation or stabilization of spacecraft. Consideration of such systems at this time is further limited to those employing cold gas. Essentially simple, such systems are relatively low in cost compared to other spacecraft subsystems, but modest cost reductions may be accomplished in their design by simplification and by the elimination of high cost materials. Furthermore, when weight and volume restrictions on spacecraft design are relaxed, cold gas systems may be used to the exclusion of such costly control systems as momentum wheels and magnetic torquing systems. Because of the inherent high-reliability of the cold-gas systems, cost savings can be effected by adding a small amount of functional redundancy and extra gas to allow doubling or tripling the operating life expectancy between refurbishments or replenishment of gas.

5.1.3.7 Environmental Control Subsystems. The Environmental Control subsystem comprises primarily the on-orbit thermal control system. Essentially passive thermal control techniques (heaters used) are used almost exclusively in unmanned spacecraft and are relatively low in cost compared to other spacecraft subsystems. However, the design of the thermal control subsystem for a complex spacecraft like the OAO-B required extensive analysis, testing and detail design of the installation of insulating materials. Appreciable cost savings can be made if such a thermal control system can be extensively simplified. Such simplification has been attained in the design of the thermal control system of the low-cost OAO discussed in this report.

5.1.4 Payload Designers Handbook

LMSC document No. A-990558, "Design Handbook for Low-Cost Space Shuttle Payloads", dtd 7 June 1971, has been prepared by LMSC as a part of the Payload Effects study effort.

In preparing the designs of low-cost payloads many factors affecting payload costs and program costs were analyzed. Methods of reducing these costs were identified and implemented in the designs and are presented in the Design Handbook.

The immediate objective of the Design Handbook is to communicate to space program planners, managers, and designers concepts for reducing the costs of space payloads in the Shuttle era. Some of the concepts are greatly at variance with past practice, and their acceptance by the leaders of the nation's space programs is essential if they are ultimately to be applied in payload design.

The new payload designs created during the Payload Effects Analysis study are preliminary and, as such, are not documented in detail in engineering documents. They, however, are described in sufficient detail for understanding and evaluation of design concepts in LMSC Engineering Memoranda which have been prepared and made available to the members of the NASA Technical Monitor Committee for the Payload Effects Analysis study.

The Design Handbook includes brief descriptions of the cost saving features of the new payload designs, presented as examples to illustrate cost-saving guidelines for future payload designs. It also includes brief summaries of Space Shuttle and Space Tug performance, environmental and interface data, to the degree that they affect the payload design concepts presented.

5.2 LOW-COST ORBITING ASTRONOMICAL OBSERVATORY (OAO) DESIGN

The designs of a low-cost Orbiting Astronomical Observatory (OAO), for launch by the Space Shuttle and by expendable launch vehicles, are derived from the OAO-B observatory and are described in the following paragraphs.

5.2.1 OAO-B, The Baseline OAO

The OAO-B was the third of the OAO series of Orbiting Astronomical Observatories. It comprised the Grumman OAO spacecraft and the NASA Goddard Experiment Package (GEP). The GEP employed a relatively fast, 38-in. (0.96 m) Cassegrain telescope with a large-aperture spectrometer to obtain high resolution spectral data from point and extended sources in the ultraviolet region of the stellar spectrum.

5.2.2 Shuttle-Launched OAO Performance and Design Requirements

The design of the low-cost OAO was preceded by preparation of a document, LMSC-A973890-A, "General Specification - Performance and Design Requirements for Low-Cost Orbiting Astronomical Observatory", dtd 5 May 1971 (Revised). The performance requirements included in the specification were the same as for the baseline OAO-B. The design was developed in general accordance with the requirements stated in this specification.

The OAO is designed to operate for one year in a circular orbit of 390 to 417 nm (722 to 772 km) altitude inclined $35^\circ \pm 1^\circ$ to the earth's equator.

The Shuttle-launched OAO will be transported to its operational orbit by the Space Shuttle. It will be checked out in the Shuttle cargo bay to insure normal functioning of all equipment before being released from the Shuttle. Equipment modules found to be malfunctioning may be replaced prior to orbit deployment with spare modules carried in the cargo bay.

5.2.3 Low-Cost Shuttle-Launched OAO Configuration

The flight configuration of the low-cost OAO is shown in Fig. 5-1. In the launch-stowed condition, the solar array paddles are folded flat against the aft face of the cylindrical equipment section and the telescope sun shield is retracted 48 in. (1.22 m), making the overall retracted length 23 ft (7.01 m). A 48 x 48 in. (1.22 x 1.22 m) opening is provided in each solar array and in the stowed position these openings are coincident and allow access to the interior of the 8 ft (2.44 m) diameter tunnel for inspection or repair prior to payload deployment from the Shuttle and for later module replacement.

5.2.4 Description of Low-Cost Shuttle-Launched OAO

5.2.4.1 Subsystems of the OAO. The subsystems comprising the OAO, both Shuttle launched and expendable-launched, and the LMSC informal Engineering Memos that describe them in detail are as follows:

Subsystem	Engineering Memo
Experiment	PE-1
Stabilization & Control	PE-2
Communications, Data Processing, & Instrumentation	PE-3
Electrical Power	PE-4
Attitude Control	PE-5
Environmental Control	PE-6
General Description of Low-Cost OAO	PE-7

Summary descriptions of all subsystems are presented in the following paragraphs. A separate Engineering Memo describing the Structures & Mechanisms Subsystem (general data was included in PE-7) has not been prepared and that subsystem is described first.

5.2.4.2 Structures & Mechanisms Subsystem. For reference purposes, the baseline OAO-B structures and mechanisms are shown in Fig. 5-2. The basic structure

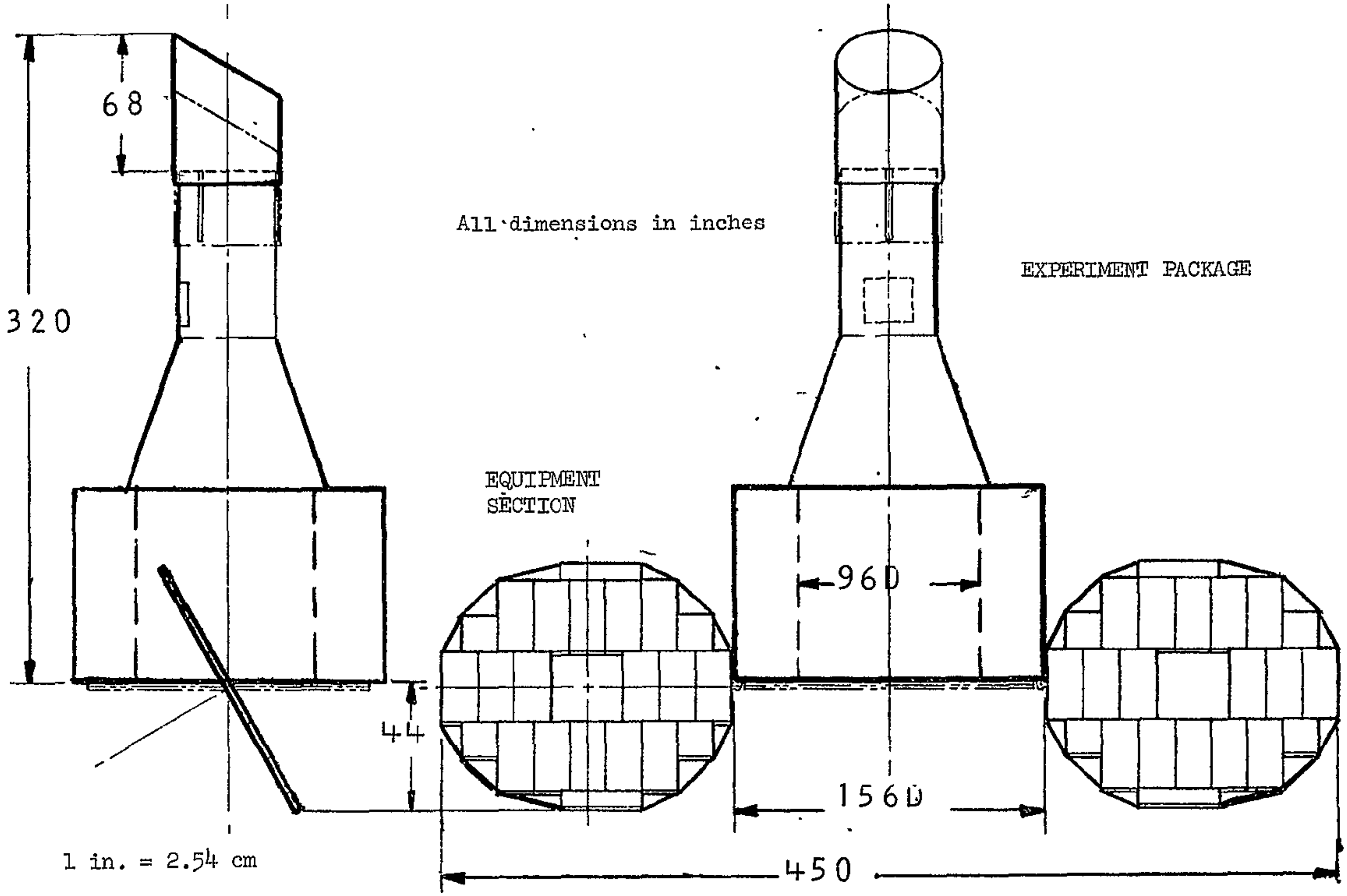


Fig. 5-1 Low-Cost OAO Configuration

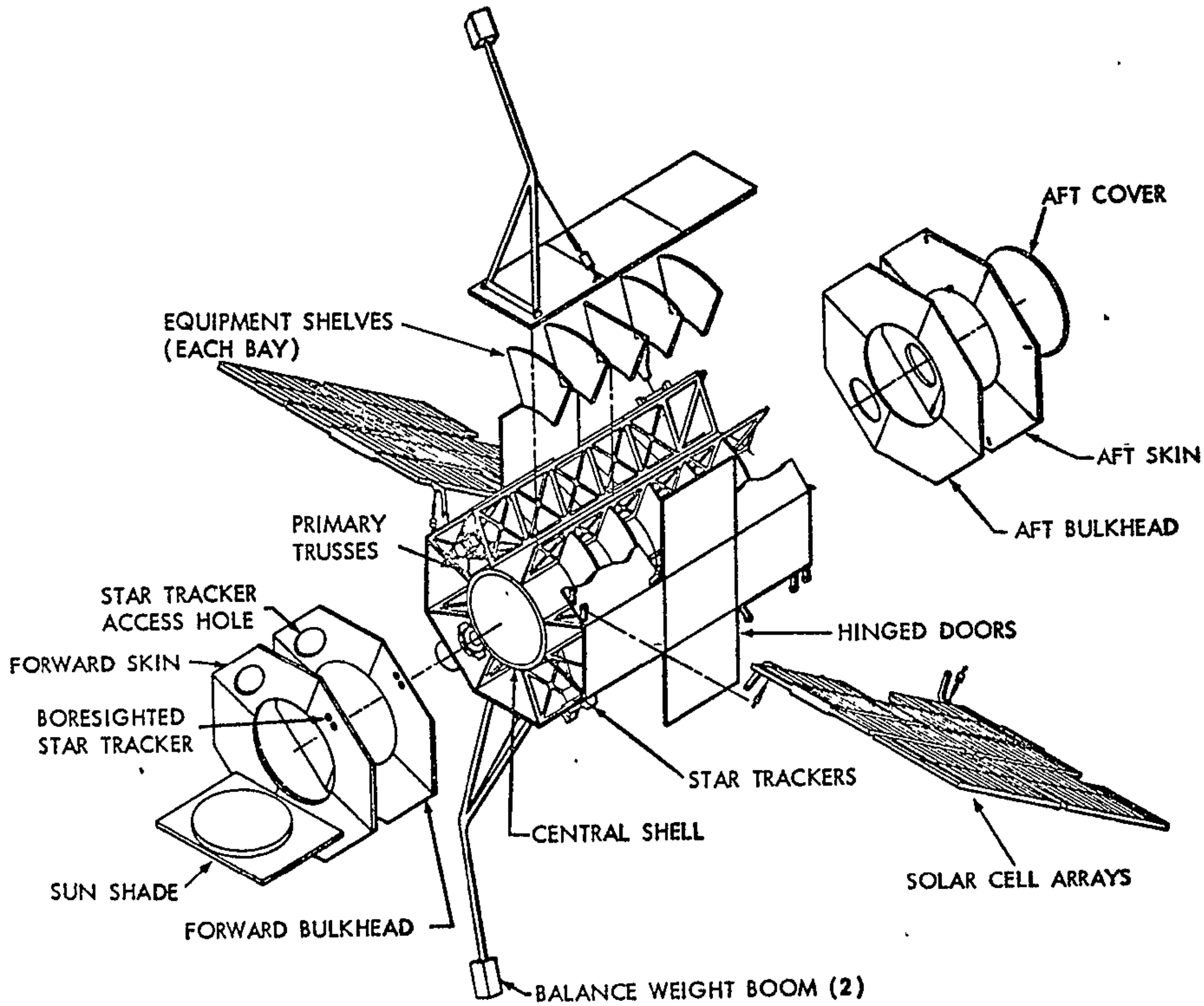


Fig. 5-2 Baseline OAO-B Structures and Mechanisms

is an octagonal cylinder approximately 7 ft (2.13 m) wide and 10 ft (3.05 m) high. It includes a central tube 48 in. (1.22 m) in diameter and 118 in. (3.0 m) long in which the experiment package is installed. The eight primary trusses are machined and chem-milled from aluminum plate. The forward and aft bulkheads and the equipment shelves form 48 equipment bays. Secondary structure is provided in each bay for the support of one or more items of equipment. Hinged doors provide access to the installed equipment.

The general configuration of the low-cost OAO equipment section structure is shown in Fig. 5-3. The equipment section is cylindrical, 13 ft (3.96 m) in diameter and 8 ft (2.44 m) long. Twenty-four equipment compartments surround an 8 ft (2.44 m) diameter by 8 ft (2.44 m) long empty volume in which ground technicians or Space Shuttle crewmen may work, installing or removing equipment modules from the equipment compartments. Modules are retained by mechanisms that are designed to be operated by a space-suited Shuttle crewman.

The equipment section structure is a riveted assembly consisting primarily of commercially-available aluminum sheet and extrusions. Factors of safety of three or greater are used to reduce design and analysis efforts and to reduce or eliminate detail static load testing.

The forward bulkhead of the equipment section is the basic reference for alignments in the low-cost OAO and machined surfaces are provided for the attachment of the experiment package and for indexing equipment modules containing attitude reference equipment such as star trackers and inertial reference units. Modules may be removed and replaced while alignment is maintained within specified tolerances.

The aft bulkhead is of sheet metal construction. Access to the work space inside the equipment section is through a 4 ft (1.22 m) square door.

The low-cost OAO equipment section structure is ruggedized and simpler than the baseline OAO-B equipment section and has far fewer piece parts. For these

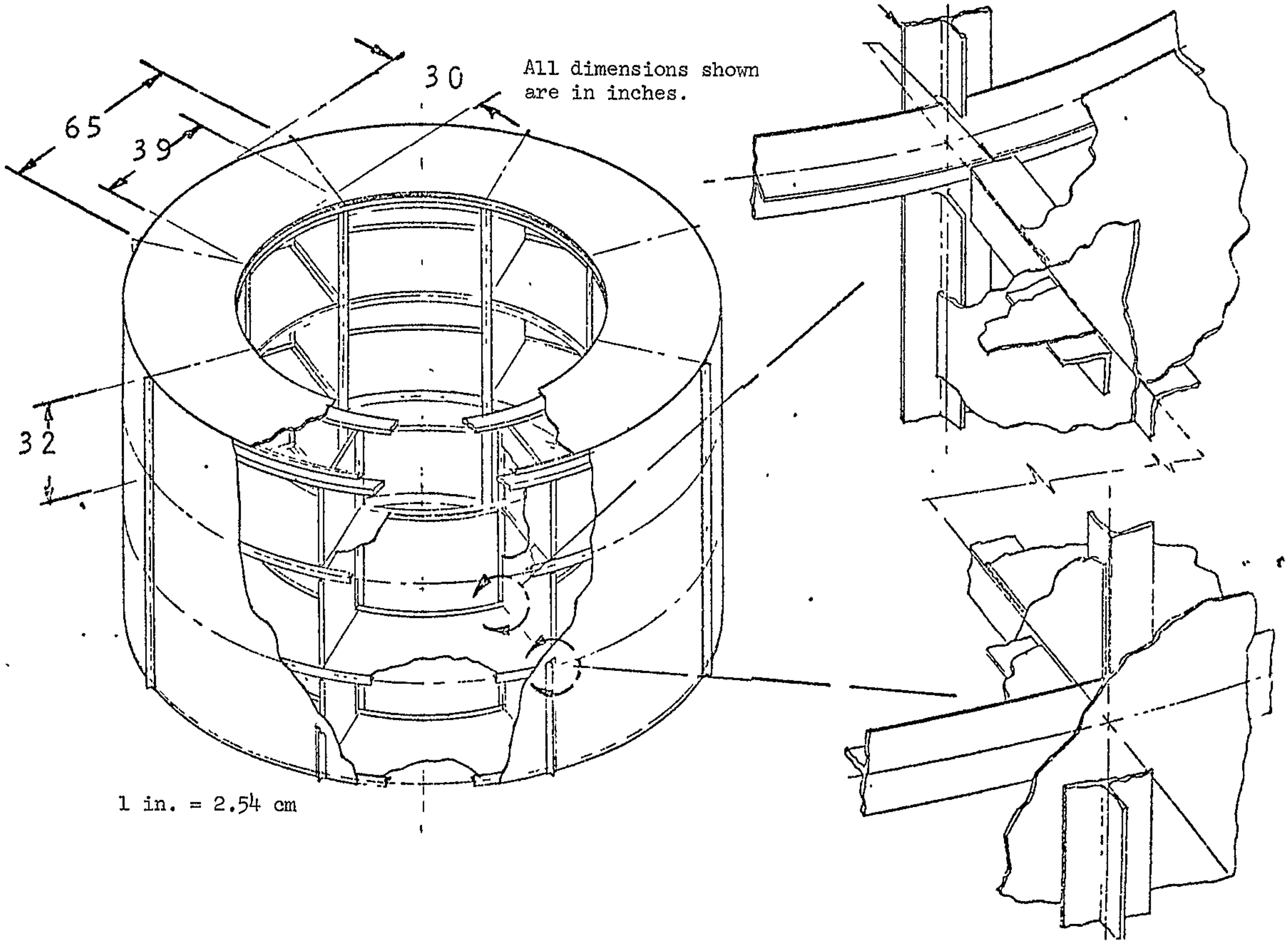


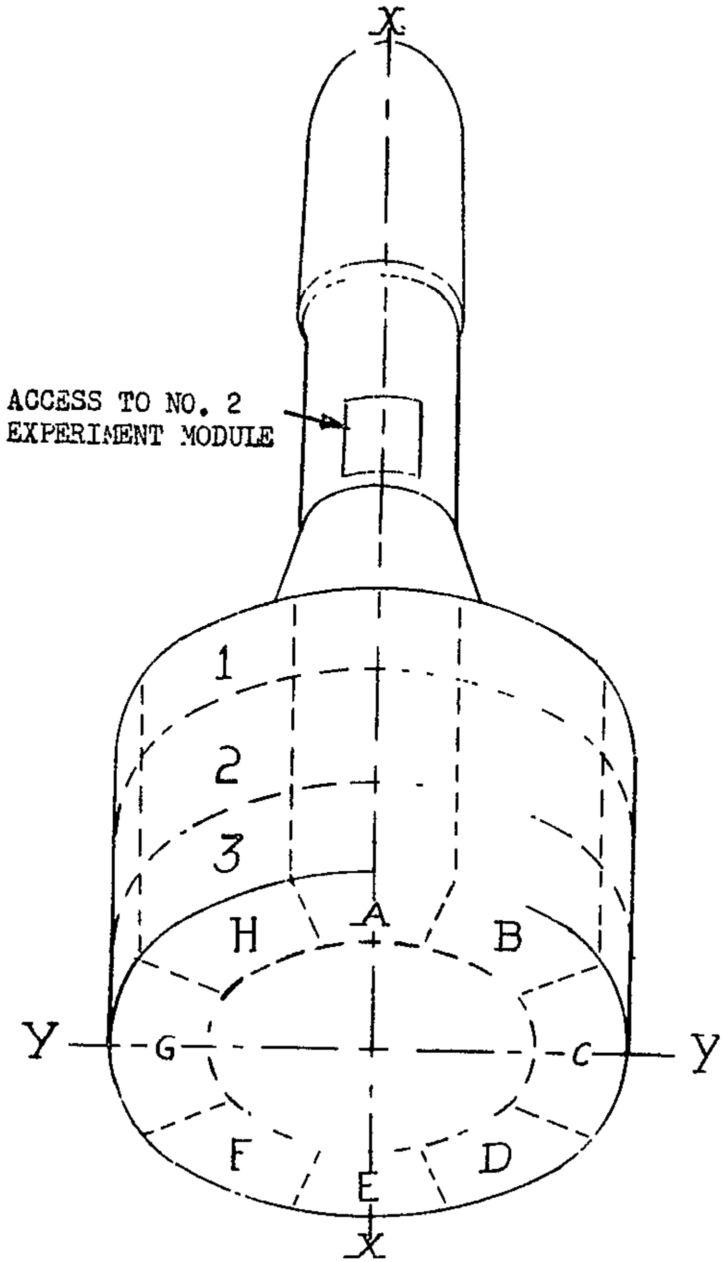
Fig. 5-3 Low-Cost OAO Equipment Section Structure

reasons, the cost of developing and manufacturing the low-cost OAO structure is significantly lower than the corresponding costs for the baseline OAO-B.

The arrangement of equipment modules in the low-cost OAO is shown in Fig. 5-4. The components comprising each module and the module weights are listed in Figs. 5-5a, 5-5b, and 5-5c. The configurations of the modules have been determined as a result of LMSC-imposed requirements, which (1) limited module sizes, weights, and unit costs; (2) provided for acceptable overall OAO weight and balance (mass distribution); and (3) dictated combination of similar functional equipment into the same module.

5.2.4.3 Experiment Subsystem. The design of a low-cost Experiment Package for the low-cost OAO was derived from the Goddard Experiment Package (GEP) of the OAO-B shown in Fig. 5-6. The GEP was designed to be installed within the central tube of the OAO-B spacecraft, and its thermal control requirements constrained and added to the complexity and cost of the thermal control design of the spacecraft. Similarly, the GEP design was complicated by the thermal environment provided by the spacecraft. Weight constraints imposed on the design of the GEP led to the use of beryllium mirrors that were more costly than mirrors of conventional optical materials. The optical properties of beryllium mirrors were unstable in 1-g, and optical calibration procedures were consequently more complex and costly than they would have been if rigid but heavier mirrors had been used. In-orbit focus control of the GEP was required to compensate for calibration uncertainty due to thermal effects and the instability of the beryllium mirrors.

The general configuration of the LMSC-designed Experiment Package for the low-cost OAO is shown in Fig. 5-7. The basic optical design of the GEP is unchanged. However, all the optical elements are fabricated from a conventional optical material, CerVit (a fused silica material) which has a very low coefficient of thermal expansion. All the mirrors and the grating are made from solid CerVit blanks having a diameter-to-thickness ratio of approximately 6 to insure adequate stiffness in a 1-g environment and to facilitate fabrication and calibration. The low-cost Experiment Package structure is an aluminum



COMPARTMENT	MODULE
A-1	EMPTY
A-2	ELECTRICAL POWER NO. 2
A-3	ATTITUDE CONTROL NO. 1
B-1	EMPTY
B-2	ELECTRICAL POWER NO. 1
B-3	EMPTY
C-1	EMPTY
C-2	EMPTY
C-3	ATTITUDE CONTROL NO. 2
D-1	STABILIZATION & CONTROL NO. 1
D-2	EXPERIMENT NO. 1
D-3	EMPTY
E-1	STABILIZATION & CONTROL NO. 3
E-2	CDPI* DATA DISTRIBUTION UNIT**
E-3	ATTITUDE CONTROL NO. 3
F-1	STABILIZATION & CONTROL NO. 2
F-2	CDPI* NO.1
F-3	EMPTY
G-1	EMPTY
G-2	CDPI NO. 2
G-3	ATTITUDE CONTROL NO. 4
H-1	EMPTY
H-2	STABILIZATION & CONTROL NO. 4
H-3	EMPTY

* Communications, Data Processing, & Instrumentation
** Fixed unit, not replaceable in orbit

Fig. 5-4 Low-Cost OAO Module Locations

SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT (LB)
ELECTRICAL	BATTERY NO. 1	o Ni-Cd Battery (3) o Power Control Unit o Power Regulator Unit o State of Charge Unit o Diode Box o Ground Power Relay o Battery Current Shunt Assy. o Module Base (20 lb) o Module Cover (25 lb) o Internal Cabling (20 lb)	Basic 390 lb 15% Cont. 58 Total 448 lb
ELECTRICAL	POWER NO. 2	o Voltage Regulator-Converter o Voltage Inverter o Power Distribution Unit o Module Base o Module Cover o Internal Cabling	Basic 170 lb 15% Cont. 26 Total 196 lb
COMMUNICATIONS, DATA PROCESSING, INSTRUMENTATION	COMPUTER NO. 1	o Digital Computer (2) o Interface & Timing Unit (2) o Cold Plate (Passive) o Module Base o Module Cover o Internal Cabling	Basic 280 lb 15% Cont. 42 Total 322 lb
COMMUNICATIONS, DATA PROCESSING, INSTRUMENTATION	COMMUNICATION EQUIPMENT NO. 2	o Tape Recorder o Narrow Band Transmitter o Wideband Transmitter o Command Receiver (2)(2 channel) o Module Base o Module Cover o Internal Cabling	Basic 90 lb 15% Cont. 14 Total 104 lb

1 lb = 0.4536 kg

Fig. 5-5a Low-Cost OAO Modules (1 of 3)

SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT (LB)
STABILIZATION & CONTROL	PRIMARY ATTITUDE REFERENCE (2 Reqd.) NO. 1 NO. 2	o Gimballed Star Tracker o GST Electronics o Inertial Reference Unit o Chassis-Alignment o Module Base o Module Cover o Internal Cabling	Basic 140 lb 15% Cont. 21 Total 161 lb
STABILIZATION & CONTROL	SECONDARY ATTITUDE REFERENCE NO. 3	o Gimballed Star Tracker o GST Electronics o Bore-Sight Star Tracker o BST Electronics o Digital Solar Aspect Sensor Electronics o Chassis - Alignment o Module Base o Module Cover o Internal Cabling	Basic 130 lb 15% Cont. 20 Total 150 lb
STABILIZATION & CONTROL -	WHEEL NO. 4	o Momentum Wheels - Fine (3) o Wheel Control Electronics o Fine Solar Aspect Sensor o FSAS Electronics o Chassis - Alignment o Module Base o Module Cover o Internal Cabling	Basic 160 lb 15% Cont. 26 Total 186 lb

1 lb = 0.4536 kg

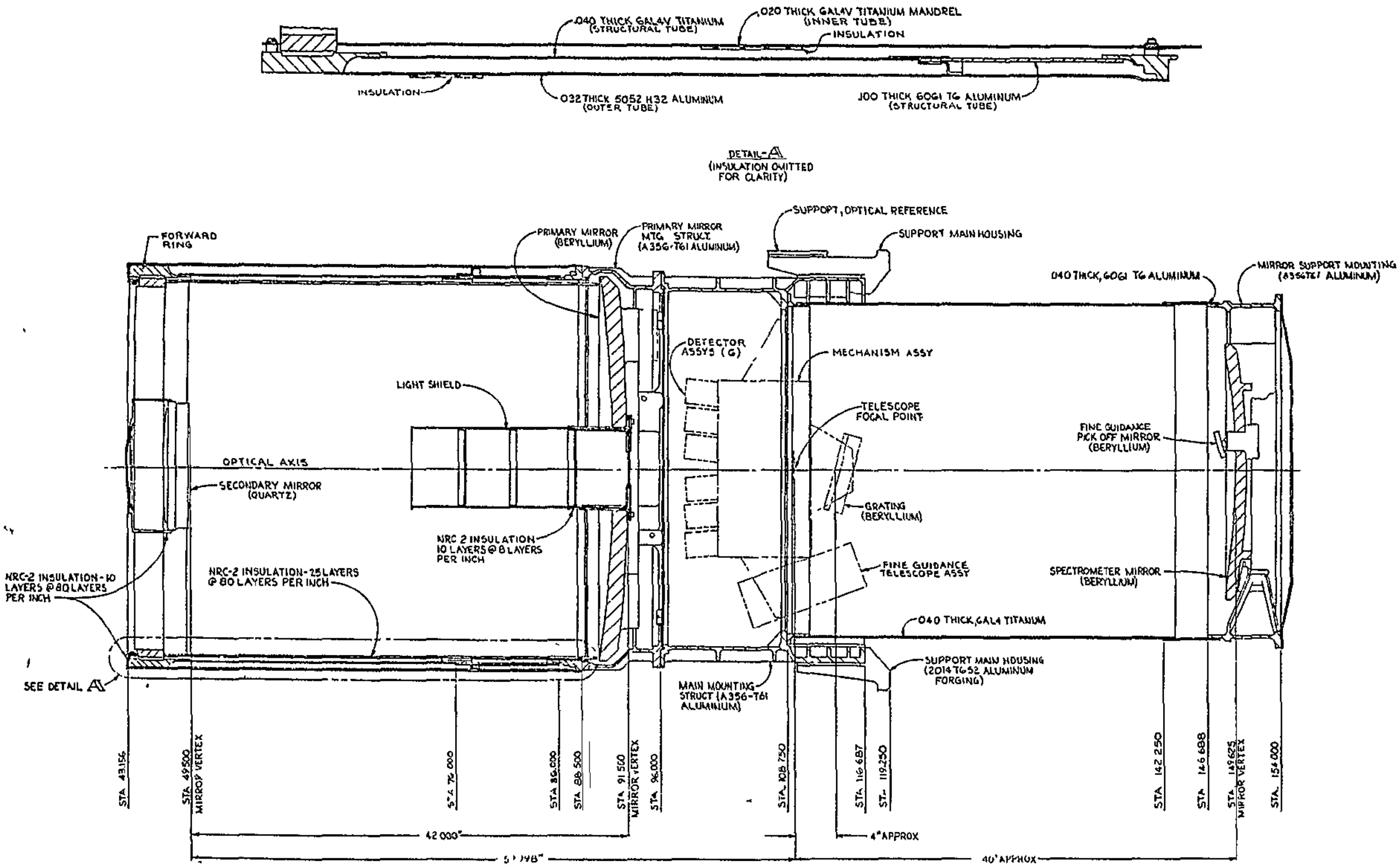
Fig. 5-5b Low-Cost OAO Modules (2 of 3)

SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT (LB)
ATTITUDE CONTROL	ATTITUDE CONTROL NO. 1 NO. 2 NO. 3 NO. 4	o Tank o Valves: Fill, Relief, Solenoid Latching, Solenoid Thrust o Regulators: High, Low o Thrust Nozzles: High (3), Low (3) o Transducers o Solar Aspect Sensor o Thruster Electronics o Sub-Chassis and Fairing o Plumbing o Module Base o Module Cover o Internal Cabling	Basic 180 lb (dry) 15% Cont. 27 Total 207 lb
EXPERIMENT	EXPERIMENT ELECTRONICS MODULE NO. 1	o Electronics Unit o Module Base o Module Cover o Internal Cabling	Basic 130 lb 20% Cont. 26 Total 156 lb
EXPERIMENT	ELECTRO-MECHANICAL NO. 2	o Focal Plane Assembly o Fine Guidance Assembly o Sub-Chassis o Module Base o Module Cover o Internal Cabling	Basic 150 lb 15% Cont. 23 Total 173 lb

Note: There is one additional module in the CDPT subsystem, the Data Distribution Unit, which is not replaceable in orbit. It has a very high reliability, being comprised of primarily passive circuitry and performs a junction box function in the Spacecraft.

1 lb = 0.4536 kg

Fig. 5-5c Low-Cost OAO Modules (3 of 3)



1 in. = 2.54 cm

Fig. 5-6 Goddard Experiment Package

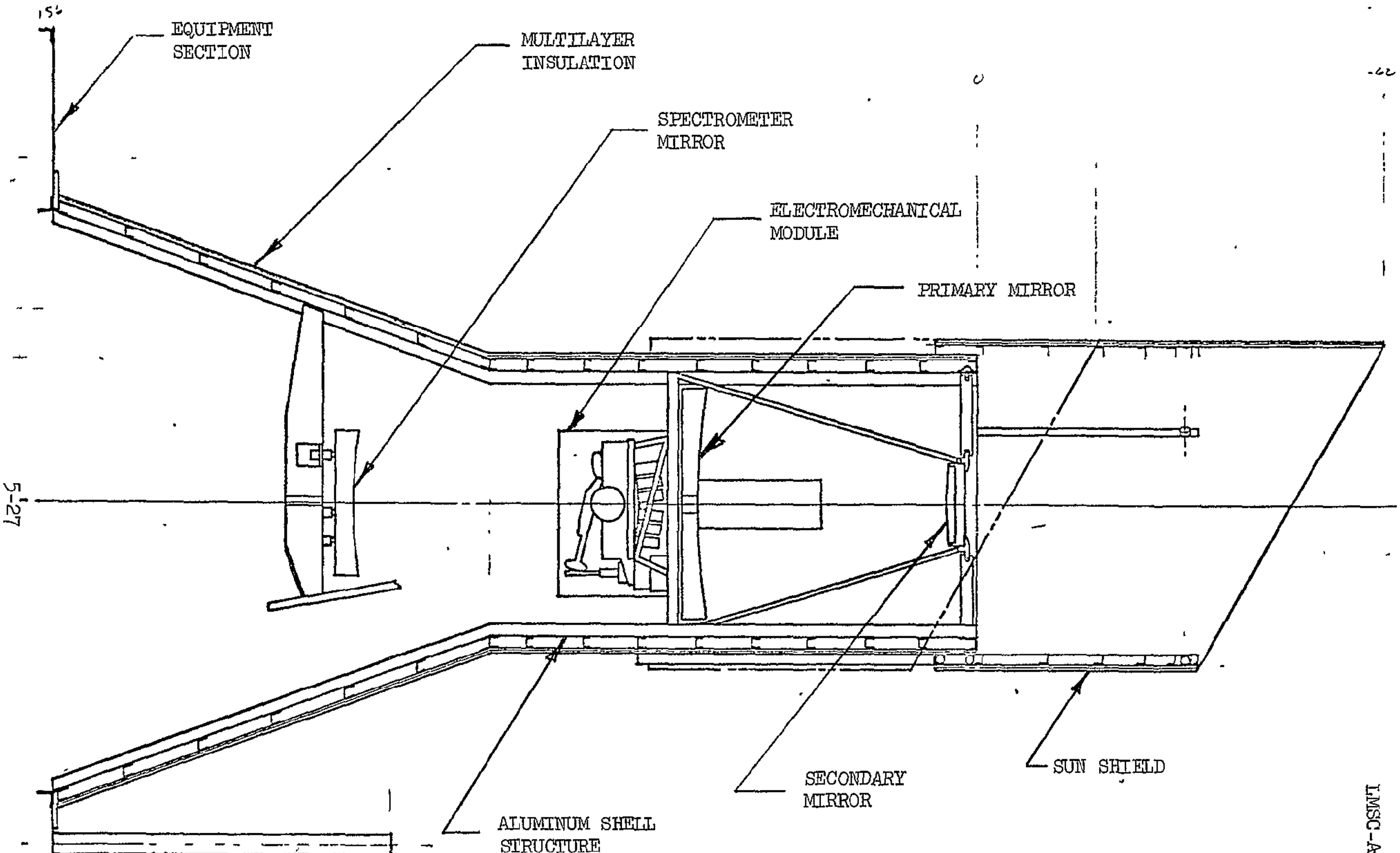


Fig. 5-7 Low-Cost OAO Experiment Package

shell stiffened by aluminum rings and longerons. All material is commercially-available aluminum.' Factors of safety of three or greater are used to (a) insure the stiffness required to preserve the alignment of the optical elements, (b) to reduce the design and analysis effort, and (c) to reduce or eliminate requirements for static load testing.

The primary mirror support is shown in Fig. 5-8. The support frame is a weldment of extruded aluminum channel designed for direct attachment to six of the internal longeron members of the shell structure. Three tangential mirror supports provide for mirror alignment.

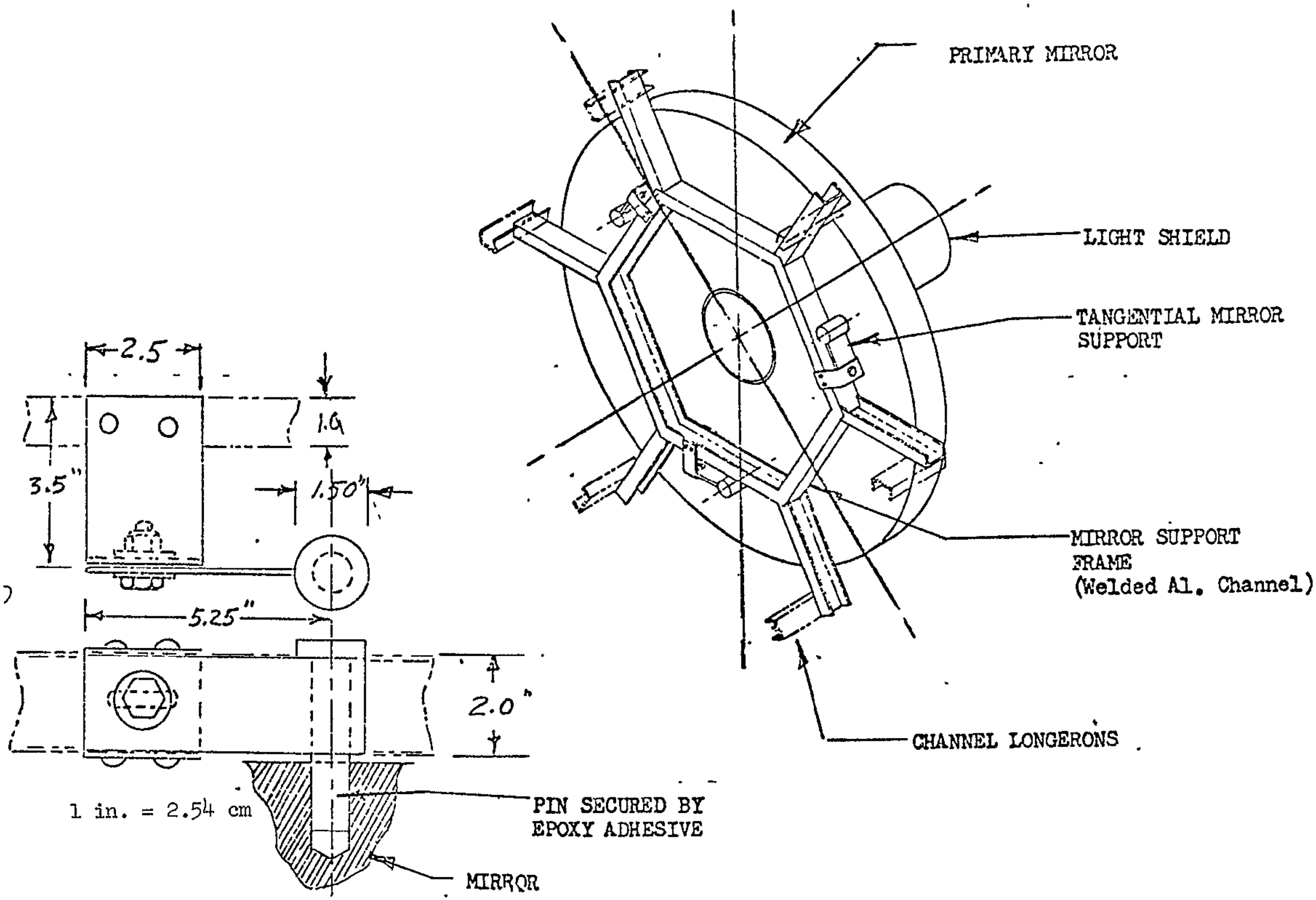
The secondary mirror is installed at the forward end of the shell structure as shown in Fig. 5-9. Support and lateral positioning of the mirror are provided by four adjustable flexure links. Alignment of the mirror is provided by three adjustment screws.

The detector and grating assembly is supported as shown in Fig. 5-10. The assembly is mounted on rails and may be removed and replaced in-orbit.

The spectrometer mirror is supported by the "tripod" beam assembly shown in Fig. 5-11. The three beams are attached to three internal longerons of the shell structure. No provision for lateral or longitudinal adjustment of the mirror is necessary. However, tilt adjustment is required and is provided by three adjustment screws.

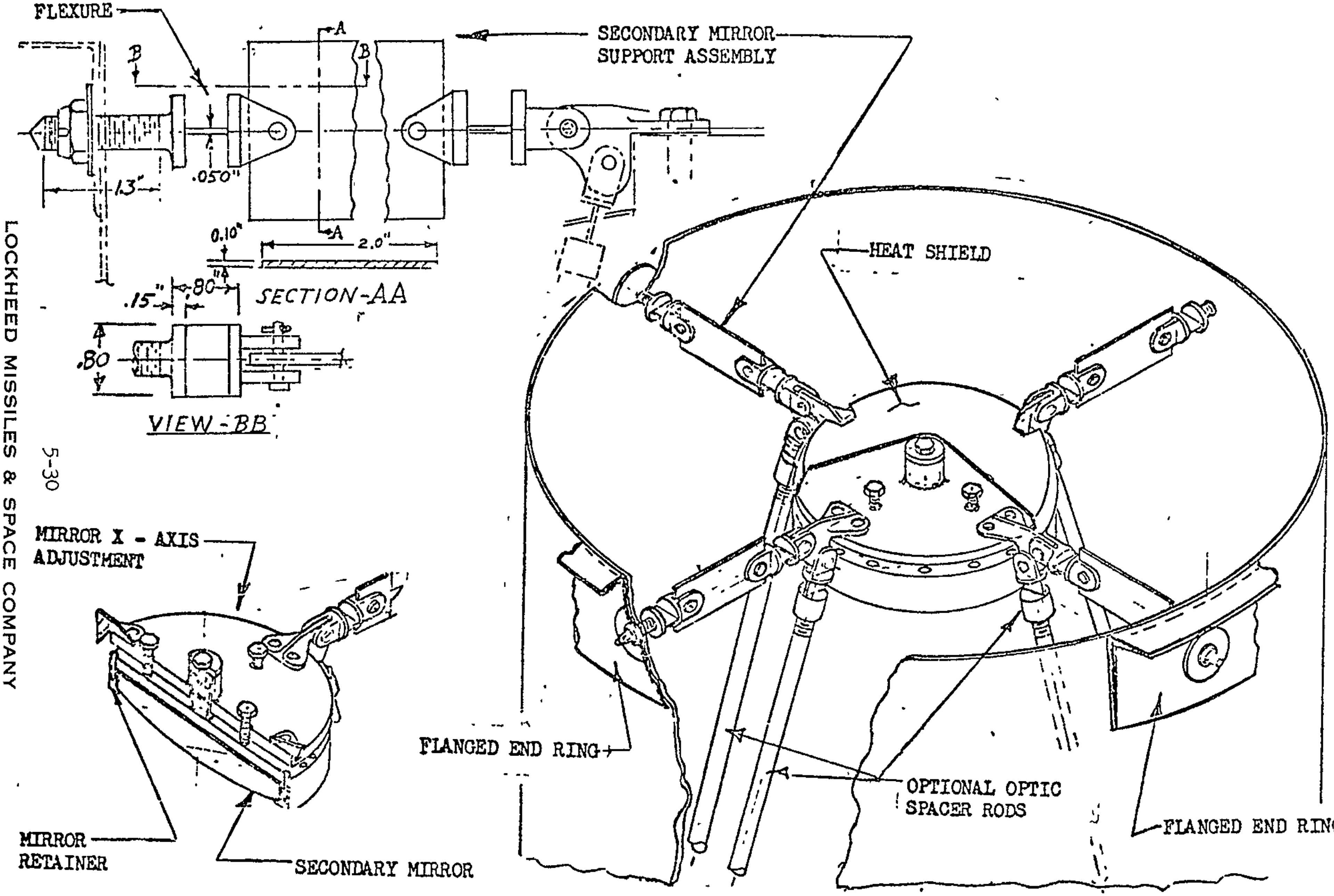
Additional cost savings in the design of the low-cost Experiment Package were achieved by the elimination of the in-orbit focus provisions, which are no longer required because of the rigidity of the structure and the virtual elimination of temperature gradients attained by isolating and insulating the experiment package.

Additional details of the low-cost OAO Experiment Package design are contained in LMSC Engineering Memo PE-1.



TANGENTIAL MIRROR
SUPPORT ASSEMBLY

Fig. 5-8 Primary Mirror Support



1 in. = 2.54 cm

Fig. 5-9 Secondary Mirror Support

LMSC-A990556

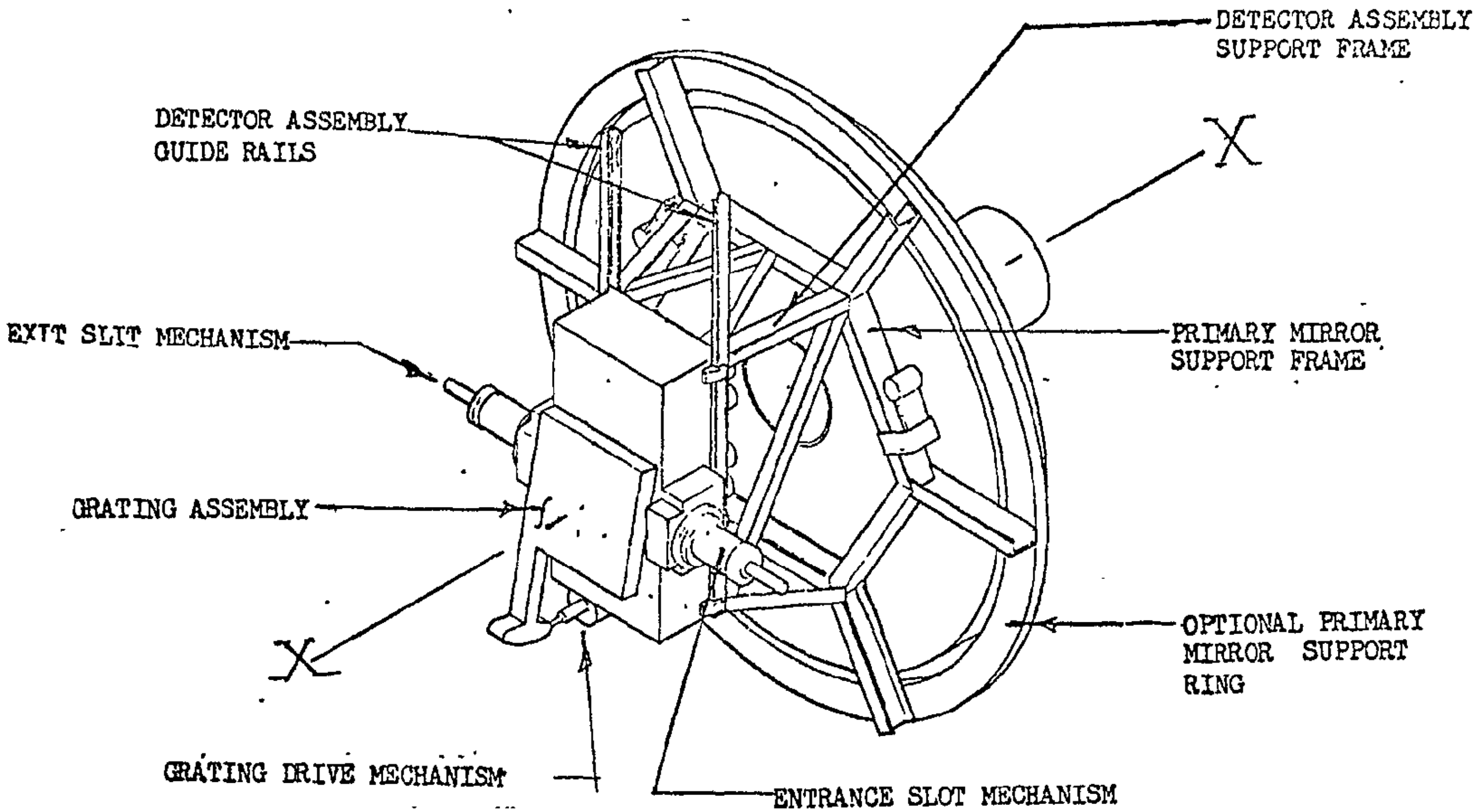


Fig. 5-10 Spectrometer Detector Assembly Mounting

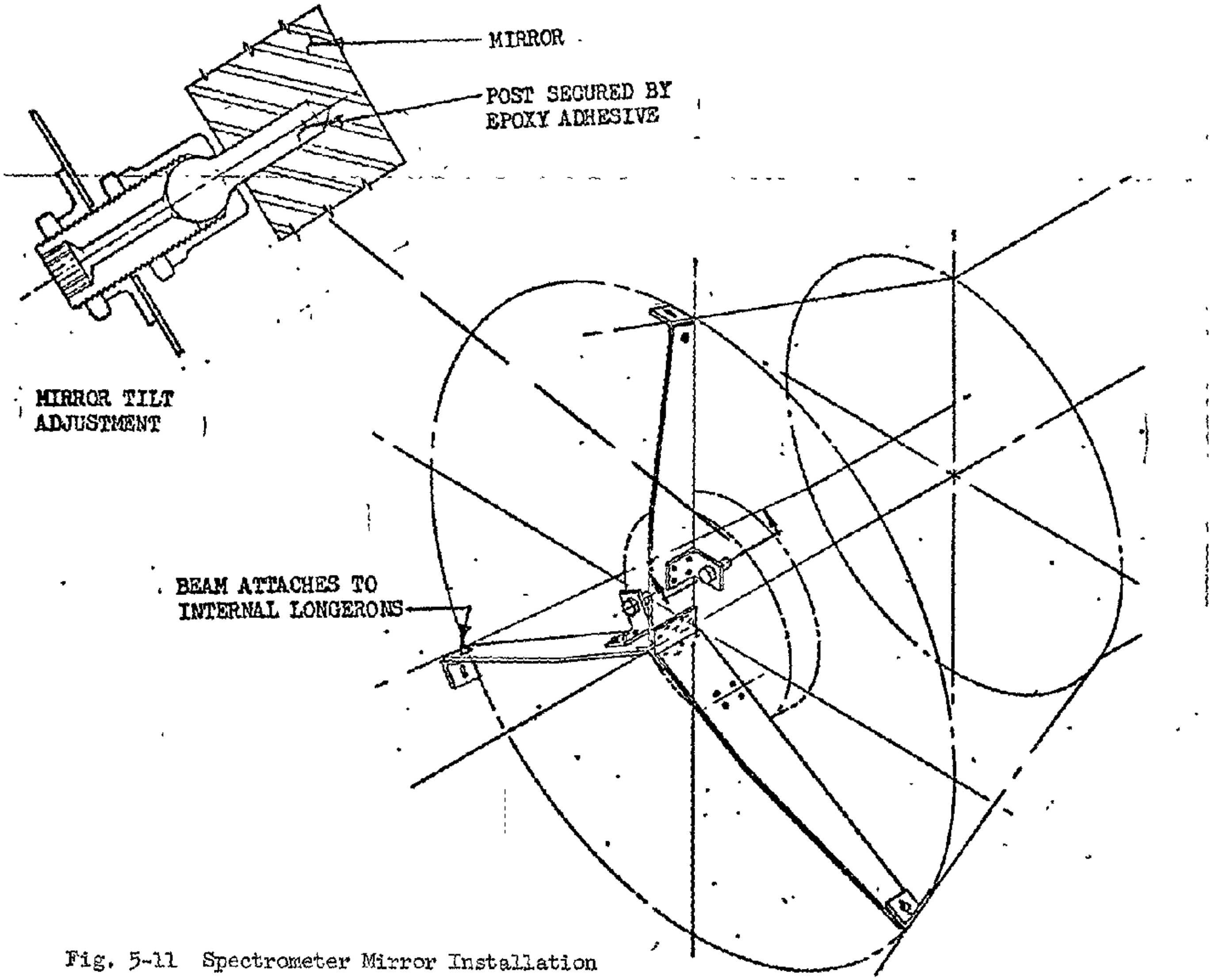


Fig. 5-11 Spectrometer Mirror Installation

5.2.4.4 Stabilization and Control Subsystem. The design of a low-cost Stabilization and Control (S&C) subsystem for the low-cost OAO is derived from the equivalent subsystem of the OAO-B. Both are described in succeeding paragraphs.

a. Baseline Stabilization & Control Functions and Hardware

The functions of the S&C subsystem of the OAO-B were as follows:

- To stabilize the spacecraft following launch vehicle separation, and establish a preselected attitude with precision.
- To orient the spacecraft, as required, to the reference attitudes dictated by the mission objectives.
- To hold the spacecraft in a reference attitude with the required accuracy over long periods of time.

To perform these functions, the OAO-B spacecraft employs gyros, solar sensors, star trackers, and magnetometers as sensors with inertia wheels, nitrogen-gas-jets, and magnetic torquers as energy devices for control and momentum unloading. With each of the sensors there is, in general, an associated electronics unit called "signal processor," and with each of the actuators there is, in general, an associated electronics unit designated a "controller." In addition, there is a logic unit, or "programmer," which sequences the S&C subsystem through its various modes of operation.

After injection into orbit, separation of the spacecraft from the Centaur results in tumbling of the spacecraft. Gyros and solar sensors detect spacecraft rate and position and produce error signals needed to actuate the gas jets and inertia wheels for stabilization. Initial stabilization could be programmed to result in one of two orientations: (1) solar paddles normal to the sun or (2) spacecraft aft end normal to the sun. From either of these orientations the spacecraft could be programmed such that control was transferred to the

gimballed star trackers. This was accomplished in one of two methods:

- <u>Star search</u> - The spacecraft orientation (three axes) was determined from solar aspect sensor and magnetometer data, while holding on the inertial reference unit. Guide Star positions were computed with respect to this spacecraft orientation. The star trackers then were programmed to search and lock on the stars by executing a star search routine.

- <u>Roll search</u> - The initial stabilization placed the aft end of the spacecraft normal to the sun. A roll search around this sunline was initiated at a given spacecraft rate. A number of guide stars were selected to form a pattern and the star trackers were programmed to the calculated guide star positions. The spacecraft then automatically switched to tracker control when the trackers acquired the guide stars simultaneously.

The spacecraft then had a stellar frame of reference to which subsequent satellite movements could be related. In addition, the inertial reference unit (IRU) now could be reset to establish its inertial three axes reference. Spacecraft maneuvers required to point the experiment were performed under tracker or IRU control. Two axes control of the satellite was transferred to the experiment for extremely accurate pointing of the spacecraft (fine control mode).

Each of the three major spacecraft axes has a large and a small inertia wheel. The large (coarse-control) wheel slews the satellite, while the small (fine-control) wheel stabilizes the satellite against small disturbing torques after the desired pointing has been attained. In the fine-control mode, the satellite can maintain its pointing direction with an accuracy of 0.1 sec by using an error signal derived from experiment optics.

To prevent the inertia wheels from becoming saturated, the gas jet system and a magnetic torquing system (a set of orthogonal electromagnetic coils) will permit momentum dumping.

In the coarse-control mode, using star trackers for error sensors, the three axes pointing accuracy should be within 1 min. with a drift rate of less than 15 sec in 50 min.

b. Low-Cost OAO Stabilization & Control Subsystem

The low-cost OAO S&C subsystem retains many of the features of the equivalent subsystem of the baseline OAO but differs from the baseline subsystem in: (1) Relying more upon on-board data processing and less on ground processing and communication; (2) eliminating equipment dedicated to alternate/backup modes, (3) reducing the amount of electronic hardware by integrating all data-processing and sequencing functions into a programmable general-purpose digital computer, and (4) eliminating primary actuator equipment whose functions can be performed efficiently by cold gas jets.

Figure 5-12 shows the functional and equipment relationships. The inertial reference unit (IRU) and two gimballed star trackers (GST) provide attitude data to the computer which combines them in a Kalman filter algorithm to compute spacecraft attitude precisely. Errors from the reference attitude produce signals to drive reaction wheels and cold gas jets to supply 3-axis stabilization and control. Primary attitude control torques are obtained by varying the speed of the reaction wheels, equivalent to the OAO "fine" wheels. Inertially-secular disturbance torques can cause the wheels to saturate (reach maximum speed); wheel desaturation is accomplished by torquing the spacecraft with the low-level cold gas jets.

Reorientation from one attitude reference to another is accomplished by a series of single axis slews derived by the on-board computer and using the high-level cold gas jets. IRU-detected attitude errors control attitude about the non-slew axes during slewing.

The Boresighted Star Tracker and Experiment Package error sensor perform the same roles as for the baseline OAO Subsystem, namely, provide a more precise pitch and yaw attitude reference than that obtained from the gimballed star trackers (GSTs).

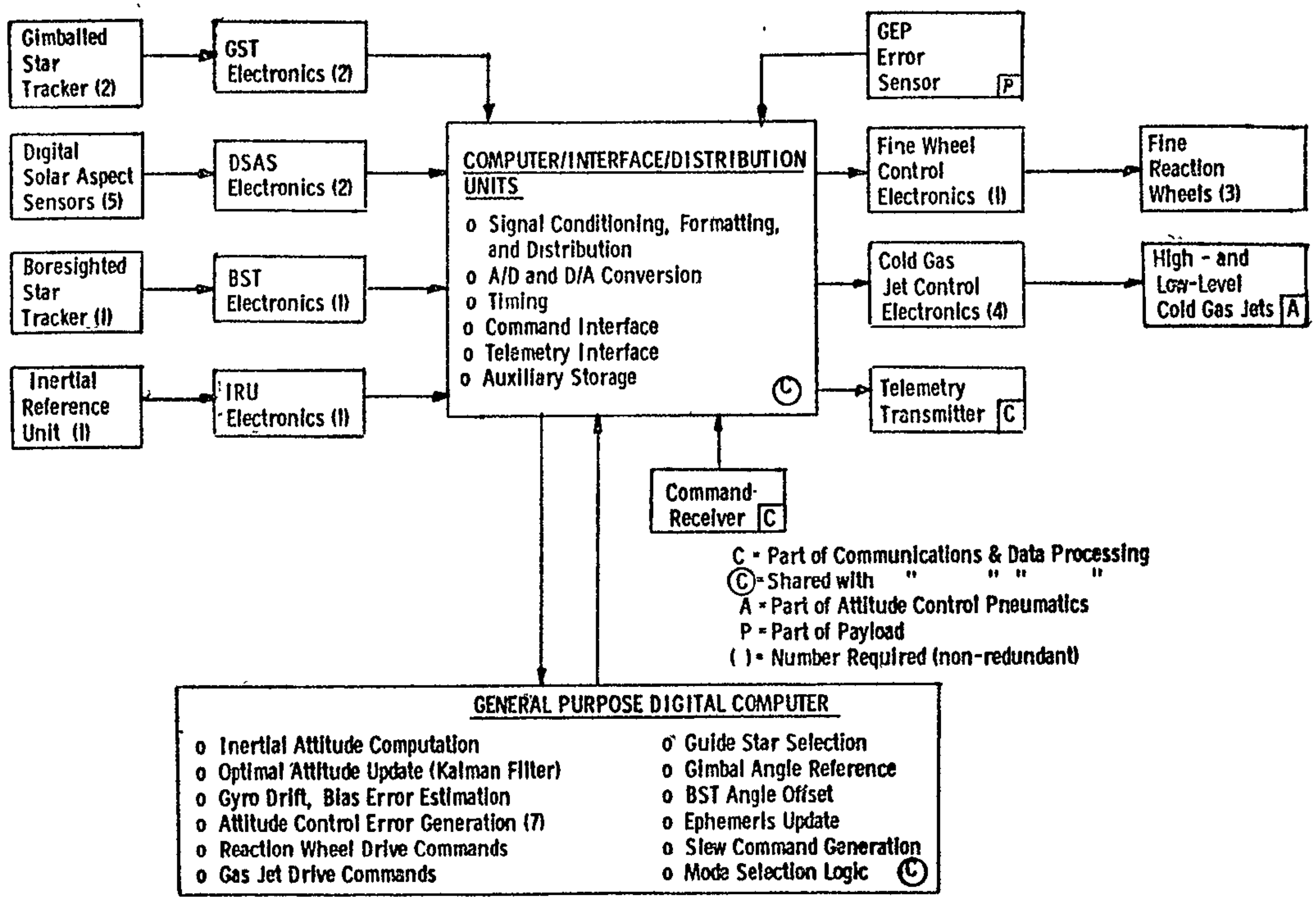


Fig. 5-12 Block Diagram - Low Cost S&C Subsystem

The principle of operation of the primary attitude determination function is to obtain accurate long-term attitude information from the GSTs and a computer stored star catalog, while the IRU gyros provide a precise short-term referenc The computer processes and mixes the information from both sources.

By continuously maintaining a record of the instantaneous attitude in the computer, the ability to acquire stars and to maneuver is considerably enhanced over the baseline OAO, and the chances of loss of star reference are diminished. This higher assurance of success in finding, holding, and reacquiring stars should reduce the level of design analysis, verification and testing efforts and lessen the need for backup modes and equipment on-board. By reducing the ground support of flight operations to periodically transmitting blocks of spacecraft reference attitudes for payload orientation, not only are ground operations minimal, but the data link support is correspondingly reduced in requisite quantity and quality, and criticality of timing. By replacing hardware with software not only are non-recurring and recurring costs reduced but the cost impact of changes which inevitably occur during development (due to inadequate performance or changed requirements) is considerably softened.

The major functional features of the low-cost OAO S&C subsystem, together with the cost reduction features of the design, are listed on Fig. 5-13. Cost reduction was accomplished by the elimination of all OAO-B equipment and operating modes not essential to compliance with the OAO-B system requirements specification; and by incorporating a general purpose computer (thus reducing the number of electronic packages needed for S&C functions) and substituting software for hardware in the processing of data.

5.2.4.5 <u>Communication, Data Processing, and Instrumentation Subsystem</u>. The design of a low-cost Communication, Data Processing, and Instrumentation (CDPI) subsystem for the low-cost OAO is derived from the CDPI subsystem of the OAO-B. Both are described in succeeding paragraphs.

FUNCTIONAL

- RETAINS ONLY ESSENTIAL OAO OPERATING MODES/EQUIPMENT
- CONTINUOUSLY COMPUTES SPACECRAFT ATTITUDE ON-BOARD
- MAKES ATTITUDE CONTROL INDEPENDENT OF LONG-TERM ATTITUDE REFERENCE
- HAS HIGH DEGREE OF AUTONOMOUS CAPABILITY
- USES SOFTWARE INSTEAD OF HARDWARE FOR SIGNAL PROCESSING/LOGIC/SEQUENCING
- RETAINS SOME DEGRADED BACKUP MODE CAPABILITIES

COST REDUCTION

- ELIMINATES TWELVE PRIMARY OAO ELECTRONICS PACKAGES
- ELIMINATES OAO COARSE MOMENTUM WHEELS AND MAGNETIC TORQUERS
- ELIMINATES OAO COARSE AND FINE SUN SENSORS
- ELIMINATES TWO GIMBALLED STAR TRACKERS
- ELIMINATES EQUIPMENT DEDICATED TO ALTERNATE MODES (RAPS)
- REDUCES RDT&E COSTS ASSOCIATED WITH DESIGN/REQUIREMENT CHANGES
- REDUCES RDT&E COSTS ASSOCIATED WITH DESIGN VERIFICATION AND TESTING
- REDUCES FLIGHT SUPPORT OPERATIONS AND EQUIPMENT INCLUDING DATA PROCESSING AND DATA LINK

Fig. 5-13 Features of Low-Cost OAO S&C Subsystem

a. Baseline OAO-B CDPI Subsystem Functions and Hardware

The functions of the OAO-B baseline CDPI subsystem are as follows:

- collect, store, and transmit experiment data to ground receiving stations
- collect, store, and transmit experiment and spacecraft equipment status information to ground receiving stations
- provide a beacon signal for ground stations to track the spacecraft
- receive, decode, and distribute real-time and stored command information from ground stations to control operation of observatory equipment
- provide observatory system timing and programming functions

Functional relationships of the CDPI subsystem are shown in Fig. 5-14. Communications equipment for receiving ground command signals and transmitting tracking signals, command verification signals, and spacecraft and experiment data consisted of four radio links:

- Radio command (154 MHz OAO-B)
- Radio tracking beacon (136 MHz)
- Wideband telemetry (400 MHz)
- Narrowband telemetry (136 MHz)

The radio command system included redundant command receivers used with two antennas in a diversity receiving system. Four receivers and two antennas for redundancy provided the radio link for ground control of the spacecraft subsystems and experiment apparatus.

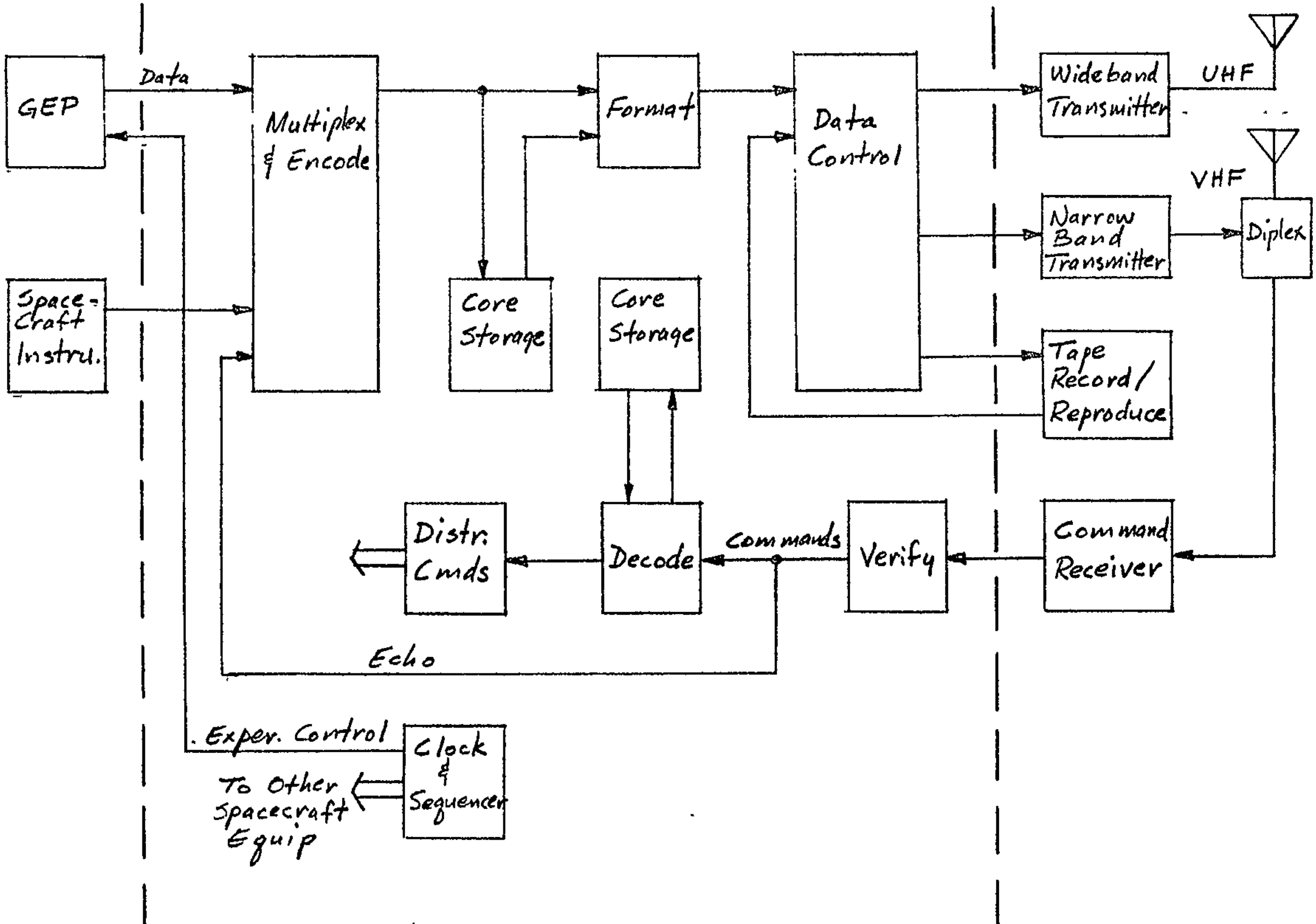


Fig. 5-14 CDPI Subsystem Functional Diagram

The radio tracking beacon supplied a continuous signal to facilitate ground tracking of the spacecraft. Ground command could select either of two identical transmitters.

The wideband telemetry link consisted of two transmitters. Ground command transmitted analog or digital data from the experimenters' units or digital data from the spacecraft. Ground command also selected the data to be transmitted: analog data were transmitted only in real time; digital data were transmitted from storage or in real time.

The narrowband telemetry link consisted of two identical transmitters. Ground command selected transmission of (1) spacecraft status data, (2) command verification, (3) command memory dump, or (4) stored status or experiment data.

From a functional standpoint, excluding the redundancy of transmitters and receivers, and antenna radiators, an antenna set operated at approximately 136 MHz with the tracking beacon and narrowband telemetry transmitter and at approximately 148 MHz with the command receivers. An ultrahigh frequency antenna set worked with the wideband telemetry at approximately 400 MHz. This antenna arrangement furnished nearly omnidirectional coverage about the observatory.

The data-processing system included circuits and storage capabilities to verify, decode, store, and distribute digital spacecraft-control and experiment commands. It also stored digital data and transmitted them from storage on ground command. The system provided timing signals for internal synchronization and for use by experiment and spacecraft equipments. The system also included spacecraft and experiment data handling equipment to accept digital data, accept and convert analog data to digital data, and assembled those data into a suitable format for storage or real-time transmission to the ground. The digital and analog gate circuitry was available for the experimenter to program data storage. Random programming of experiment data into storage was permissible. Real-time transmission of experiment data backed-up the data-storage system. During real-time transmission, data sources were sampled in a cyclic time sequence controlled by a real-time programmer included as part of the experiment data-handling equipment.

The command system contained circuits for decoding, distributing, and storing commands received from the ground-command station over the radio command receiver system. The command decoder handled the real-time commands to be executed immediately after verification in the spacecraft, and the stored commands to be placed in storage (after onboard verification) for execution at a later time. In addition, all commands could be retransmitted to the ground. Both real-time and stored commands could be transmitted over the narrowband telemetry link, and command memory could be transmitted over the wideband telemetry link to permit ground verification. Stored commands were programmable and could be executed in small time-address increments. If two or more commands had the same execution time code, they were executed within the same increment in the same order received.

A system clock provided the timing signals or pulses required by the observatory and synchronized all data words and command words.

Encoded analog data and data originally generated in binary form, was assembled into a suitable format for storage or real-time transmission to the ground over the narrowband-transmitter system.

The equipment required to mechanize the baseline OAO-B CDPI subsystem is shown later (compared with the low-cost CDPI subsystem) on Fig. 5-16 The Experiment and Spacecraft Data Handling equipment read, encoded, and assembled digital, bi-level, and analog data signals into a format suitable for transmission or storage.

The Primary Processor and Data Storage unit and the Programmer and Star Tracker Signal Controller performed the following functions:

- Verify the accuracy of ground commands
- Provide command storage
- Execute stored or real-time commands
- Supply all basic signals for the spacecraft
- Provide data storage

- Control the sequencing of operations
- Control the mode of operation of the gimballed star trackers

b. Low-Cost OAO Communication, Data Processing and Instrumentation (CDPI) Subsystem

Functional elements between the dashed lines of Fig. 5-14 comprised the Data Processing System (DPS). Two possible approaches to the DPS design were (1) separate command decoders, programmers, multiplexers, A-D converters and core storage as in the Baseline OAO-B CDPI Subsystem design, and (2) use of a central computer in an integrated data processing system. The low-cost Stabilization and Control subsystem requires the use of a digital computer to perform control system computations. Because the computer contains storage elements and provides great flexibility in sequencing, formatting, decoding and multiplexing functions; it can be utilized to great advantage also as the central unit in the DPS. With this approach, inflexible hardware was eliminated by the use of computer software programs which can be changed readily to suit evolutionary experiment and spacecraft operational changes. Therefore, this approach has been adopted in the low-cost CDPI subsystem design.

The low-cost OAO CDPI subsystem block diagram is presented in Fig. 5-15.

The Data Processing System (DPS) consists of a Data Distribution Unit (DDU), Interface and Timing Unit (ITU), and redundant computers. The integrated DPS performs all the data handling, timing, sequencing, command decoding, S&C subsystem computation, and data storage functions. The DDU and ITU provide the interfaces with all other spacecraft and experiment equipment. The DDU is essentially a junction box plus submultiplexers for instrumentation. The ITU provides timing, multiplexing, analog-to-digital conversion, and is the primary computer/spacecraft interface. The computer provides the computation and storage functions.

The tape recorder is operated under control of the computer via the DDU and records data at 1042 bps and plays back at 66.7 kbps. Output of the tape

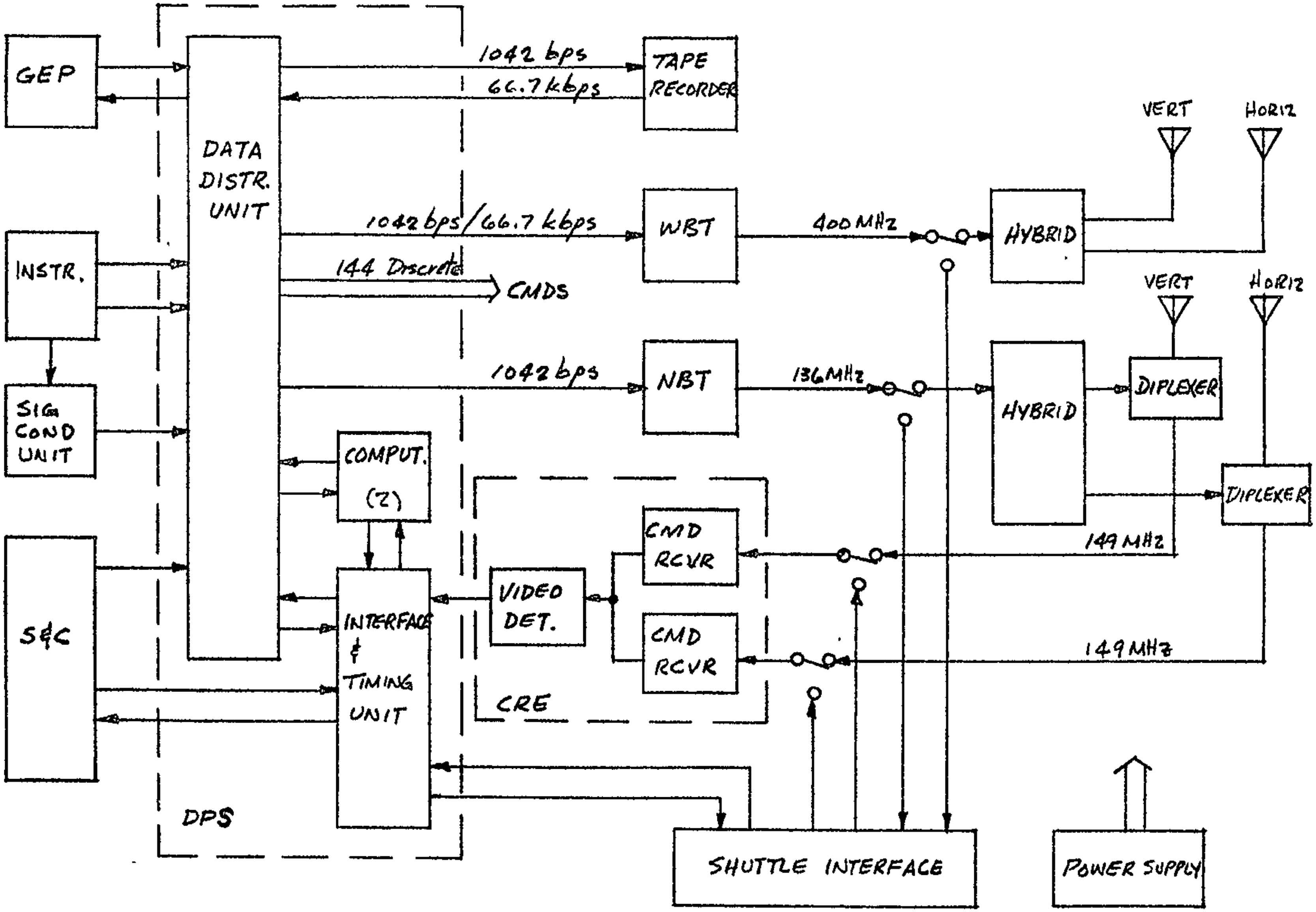


Fig. 5-15 Low-Cost OAO CDPI Subsystem

recorder is routed via the DDU to the WBT for transmission to the ground stations. Recording capability is 43.2×10^6 bits for a maximum recording time of 12 hours.

The Wideband Transmitter (WBT) accepts modulation data at either 1042 bps or 66.7 kbps from the DDU and transmits it on a carrier frequency at 400 MHz through two orthogonal, linearly polarized antennas. The transmitter power output is equally divided by the hybrid to feed the two antennas.

The Narrowband Transmitter (NBT) accepts modulation data at 1042 bps from the DDU and transmits it as PCM/PM on a carrier frequency of 136-137 MHz via the two orthogonal, linearly polarized VHF antennas. The hybrid divides the transmitter power equally between the two antennas; the diplexers provide isolation between the telemetry and command frequencies which share the same antenna.

Two AM command receivers are used to provide polarization diversity reception which assures command reception regardless of spacecraft attitude. The receiver outputs are combined for a common video detector which demodulates the FSK subcarriers to recover the PCM command words. The digital command words are then presented to the computer via the ITU for verification and execution or storage.

Interface with the Space Shuttle is by means of coaxial RF lines and RF switches which enable exercise of the payload, including telemetry transmitters and command receivers. The Shuttle interface equipment may either provide onboard checkout and diagnostics or serve as a relay station to the ground data acquisition network.

The equipment of the baseline OAO-B CDPI subsystem and the low-cost OAO CDPI subsystem are compared in Fig. 5-16.

The digital computer, ITU, and DDU of the low-cost OAO CDPI subsystem replace seven major electronic units of the OAO-B CDPI subsystem, and replace numerous electronic units of the OAO-B Stabilization and Control subsystem. The

Item	Baseline OAO-B CDPI	Low-Cost OAO CDPI
Command Receiver	4	4
Radio Tracking Beacon	2	-
Narrow-Band Transmitter	2	1
Wide-Band Transmitter	2	1
Diplexer	2	2
Hybrid Junctions	2	2
Antenna	4	4
Primary Processor & Data Storage	1	-
Programmer & St Signal Controller	1	-
Experiment Data Handling Equipment	1	-
Spacecraft Data Handling Equipment	1	-
Spacecraft Systems Controller	1	-
Signal Conditioning Unit	1	-
Auxiliary Command Memory	1	-
Instrumentation (Set)	1	1
Digital Computer	-	2
Interface & Timing Unit (ITU)	-	2
Data Distribution Unit (DDU)	-	1

Fig. 5-16 OAO-B and Low-Cost OAO CDPI Subsystem Equipment Comparison

incorporation of the computer is the most significant cost-saving change made in the low-cost CDPI subsystem. Two computers and two ITUs in time-share redundancy are required to meet the reliability goal for the subsystem.

5.2.4.6 Electrical Power Subsystem. The design of a low-cost Electrical Power Subsystem for the low-cost OAO is derived from the Electrical Power subsystem of the OAO-B. Both are described in succeeding paragraphs.

a. Baseline OAO Electrical Power Subsystem Functions and Hardware

The major equipments for the baseline OAO-B Electrical Power Subsystem (EPS) are shown in Fig. 5-17. A silicon solar cell array and three secondary nickel cadmium batteries supplied continuous power to the system loads during orbital light and dark periods, respectively. A dc-dc converter and a dc-ac inverter furnished power to each of the spacecraft subsystem equipments requiring regulated dc or ac input voltages. The nominal unregulated bus voltage was 30 volts and varied from 24 volts to 35 volts, depending on the particular OAO operating mode.

The Electrical Power Subsystem comprised the following major equipments:

(1) Solar Cell Array. The solar cell array converted solar energy to electrical energy. The array consisted of four main and four auxiliary paddles with solar cells on one side only. The cells were diffused-junction N-on-P silicon (1 cm x 2 cm) with 6 mil (0.15 mm) cover-glasses. The array contained 53,992 cells, and the operating voltage was 0 - 50 VDC (approximate).

(2) Voltage Regulator and Converter. The voltage regulator and converter converted unregulated (23v to 35v) D.C. voltage to five regulated D.C. voltage levels (+28v, -28v, +18v, +10v, -10v).

(3) Voltage Inverter. The voltage inverter converted unregulated (23v to 34v) D.C. voltage to regulated AC voltage (115v ∅, 115v 2∅, and 115v 3∅).

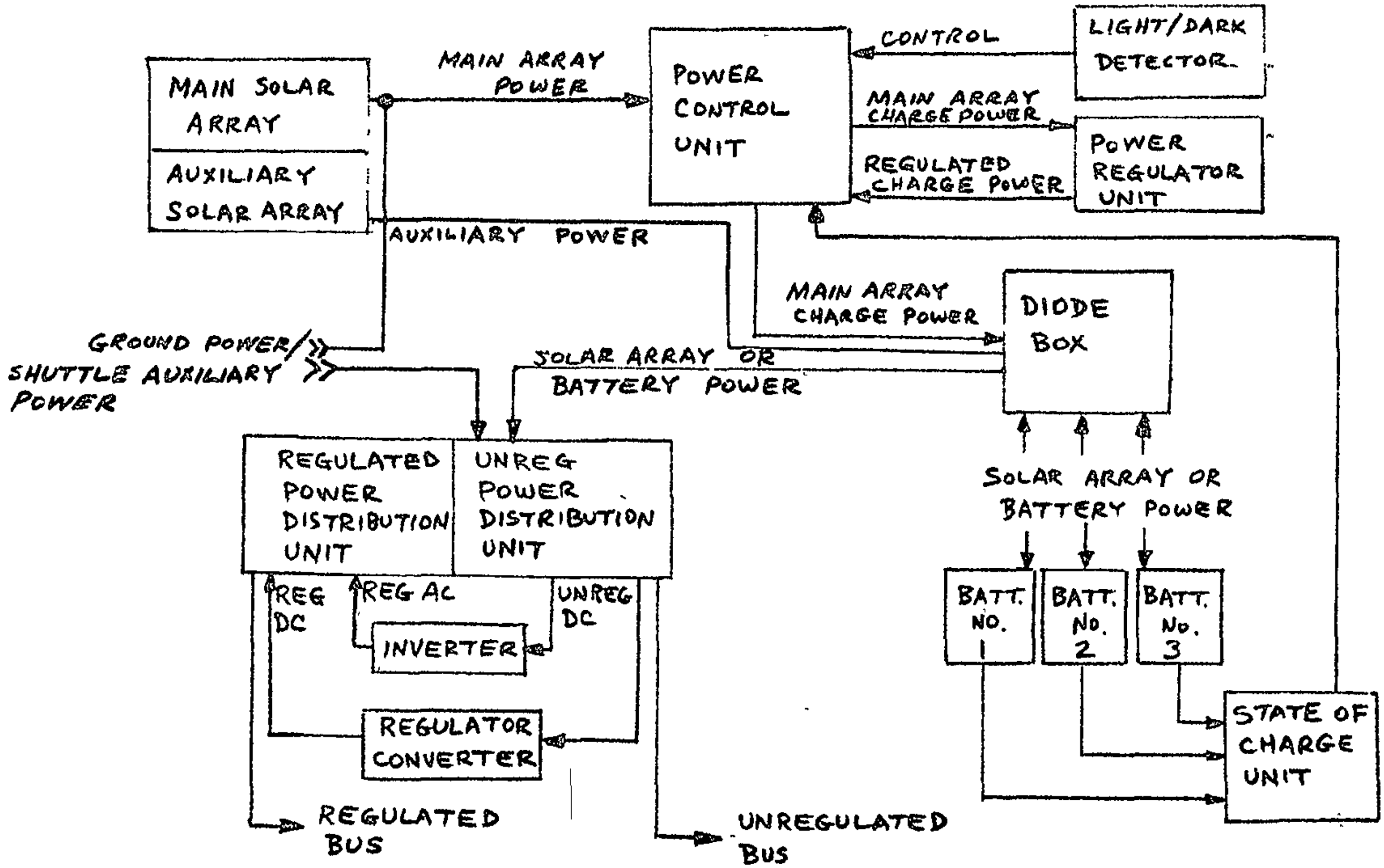


Fig. 5-17 OAO-B Electrical Power Subsystem Block Diagram

(4) NiCd Batteries. The NiCd batteries supplied power to the spacecraft during dark periods and aided in supplying peak loads during light periods. The batteries supplied power at 23 volts to 29 volts.

(5) Power Controller Unit (PCU). The power controller unit controlled individual battery charging, provided solar array power to the unregulated bus, provided ground command capability, and provided bi-level telemetry information on the Electrical Power Subsystem status.

(6) Power Regulator Unit (PRU). The power regulator unit regulated battery charging voltage to the level indicated by the PCU, provided a signal to the PCU statusing the main regulator, and provided an alternate source of regulated voltage following main regulator failure.

(7) Diode Box. The diode box supplied power to the unregulated bus during both light and dark periods, provided diode isolation between batteries and transfered battery charge current from the charge bus to the batteries.

(8) Power Distribution Panel (PDU), Regulated and Unregulated. The power distribution panel provided the central distribution point for all spacecraft power, and housed all power line fuses, current sensors, relays, etc.

(9) State of Charge Unit (SOCU). The state of charge unit monitored battery charge and discharge operation using ampere-hour integrators and adhydrodes, and telemetered data. The unit was not normally in the primary battery charging loop and required two enabling operations to provide an input to the PCU.

(10) Battery Current Shunt Assy. The battery current shunt assembly measured battery charge/discharge currents and provided proportional outputs to the SOCU Amp-Hr integrators.

The energy requirements imposed on the EPS were directly dependent on the operational mode of the satellite. The energy supply capabilities of the EPS depended on the satellite orientation with respect to the sun, because of the fixed array configuration. The satellite energy status was monitored and the operating modes were controlled in order to prevent stored energy falling below predetermined minimum requirements. An operating mode, sunbathing, could be initiated if the system energy state was running down. In sunbathing the satellite was oriented to position the fixed solar array normal to the sun's rays. The satellite was powered down and all excess electrical power went into returning the batteries to full charge.

In general, the EPS operated as follows:

(1) The satellite entered the sunlight, with the batteries having supplied all energy requirements during the previous dark period.

(2) Battery charging commenced in the shunt mode (direct connection of the solar array to all batteries in parallel).

(3) When a battery limiting condition was reached, the charge mode was changed from shunt to regulated mode, and a taper charge was then established on that battery for the remainder of the light period.

(4) The satellite entered the dark period, battery charging ceased, and the batteries supplied the spacecraft load.

b. Low-Cost OAO Electrical Power Subsystem

The block diagram of the baseline OAO-B Electrical Power subsystem, Fig. 5-17, is also applicable to the low-cost OAO Electrical Power Subsystem; and the descriptions of the equipment and operation of the baseline OAO-B EPS are, in general, applicable to the low-cost OAO EPS. However, significant

changes have been made in the design of the solar array to reduce costs. The low-cost OAO solar array shown in Figs. 5-18 and 5-19, consists of two rigid paddles in contrast to the two articulated paddles, each consisting of four panels, of the baseline OAO-B.

Each of the two paddles of the low-cost OAO solar array comprises sixteen 18 in. x 40 in. (46 cm x 102 cm) panels, four 18 in. x 20 in. (46 cm x 51 cm) panels, and two 9 in. by 40 in. (23 cm x 102 cm) panels all bolted to an aluminum structural frame. The panels are made of aluminum sheet stiffened by aluminum angles flush-riveted to one side of the sheet. The solar cells are cemented to the other side in appropriate series-parallel grouping. Solar paddle structures constructed in this way are heavier but considerably less costly than paddle structures of honeycomb or other lightweight composites or materials. 2 cm x 2 cm solar cells (rather than 1 cm x 2 cm cells) are used i the solar array of the low-cost OAO with consequent savings in assembly labor. 20 mil (0.51 mm) glass covers are used to minimize losses due to breakage in fabrication and installation of the covers. Larger cells, 2 cm x 4 cm, should be used when available to further reduce solar array assembly costs.

The most significant factor contributing to the lower cost of the low-cost OAO solar array is the enlargement of the acceptance limits on a production lot of cells, thus reducing the cost per cell. A lot of solar cells is first inspected for mechanical defects and approximately 25 percent are rejected. The remaining cells are then tested to determine a current-rating for each cell. The current rating distribution of a typical lot of cells is presented in Fig. 5-20. It has been the practice in space programs to reject approximately one-third of the lowest-rated cells (all cells below 132 ma of the example lot) to minimize the number of cells (and hence the weight) of a solar array with a given power output.

For the low-cost OAO solar array, weight is not a constraint and approximately 97.5 percent of the functional cells of a manufacturing lot (all cells above 110 ma of the example lot) may be used.

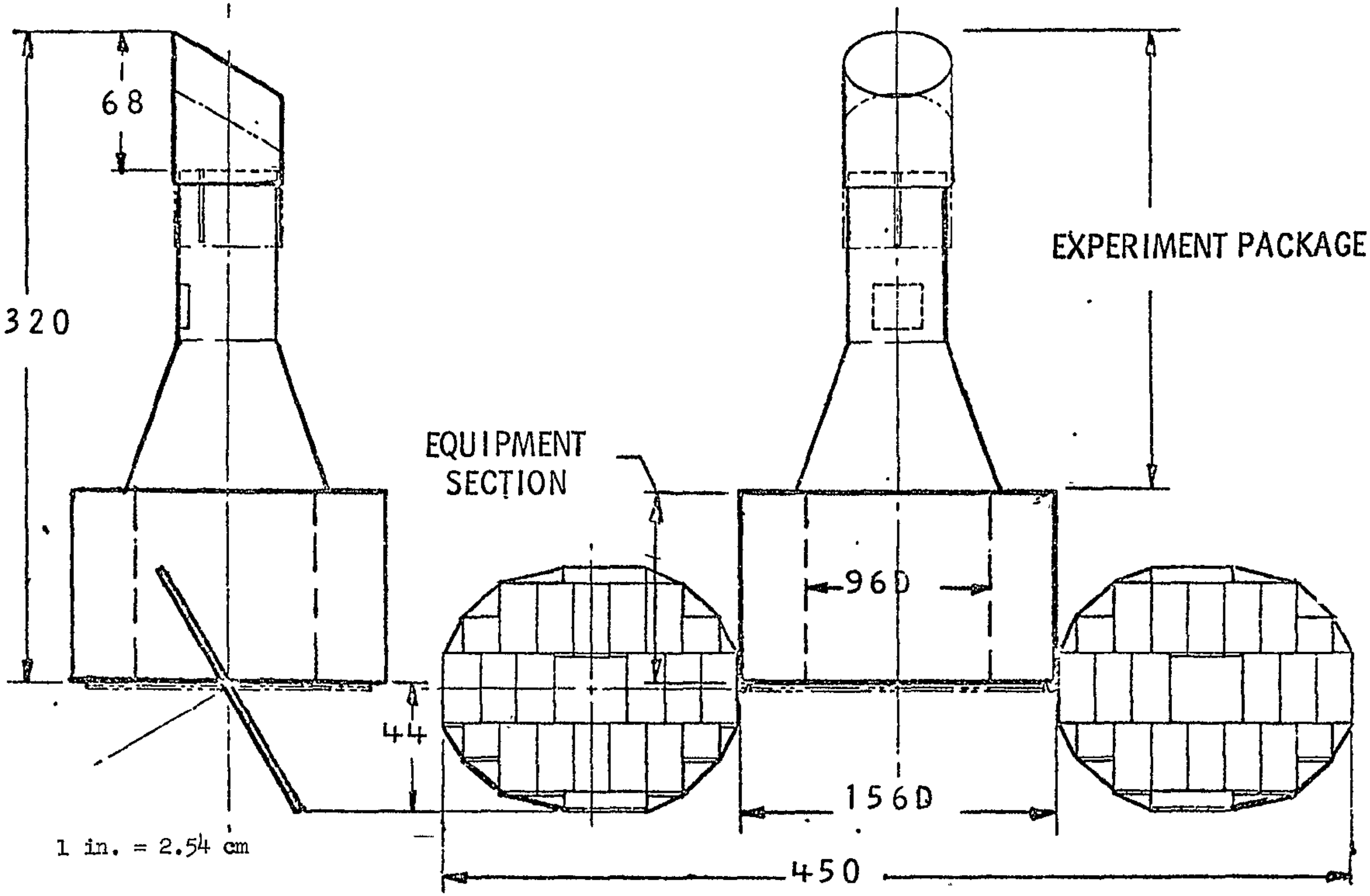


Fig. 5-18 Low-Cost OAO Solar Array Paddle Arrangement

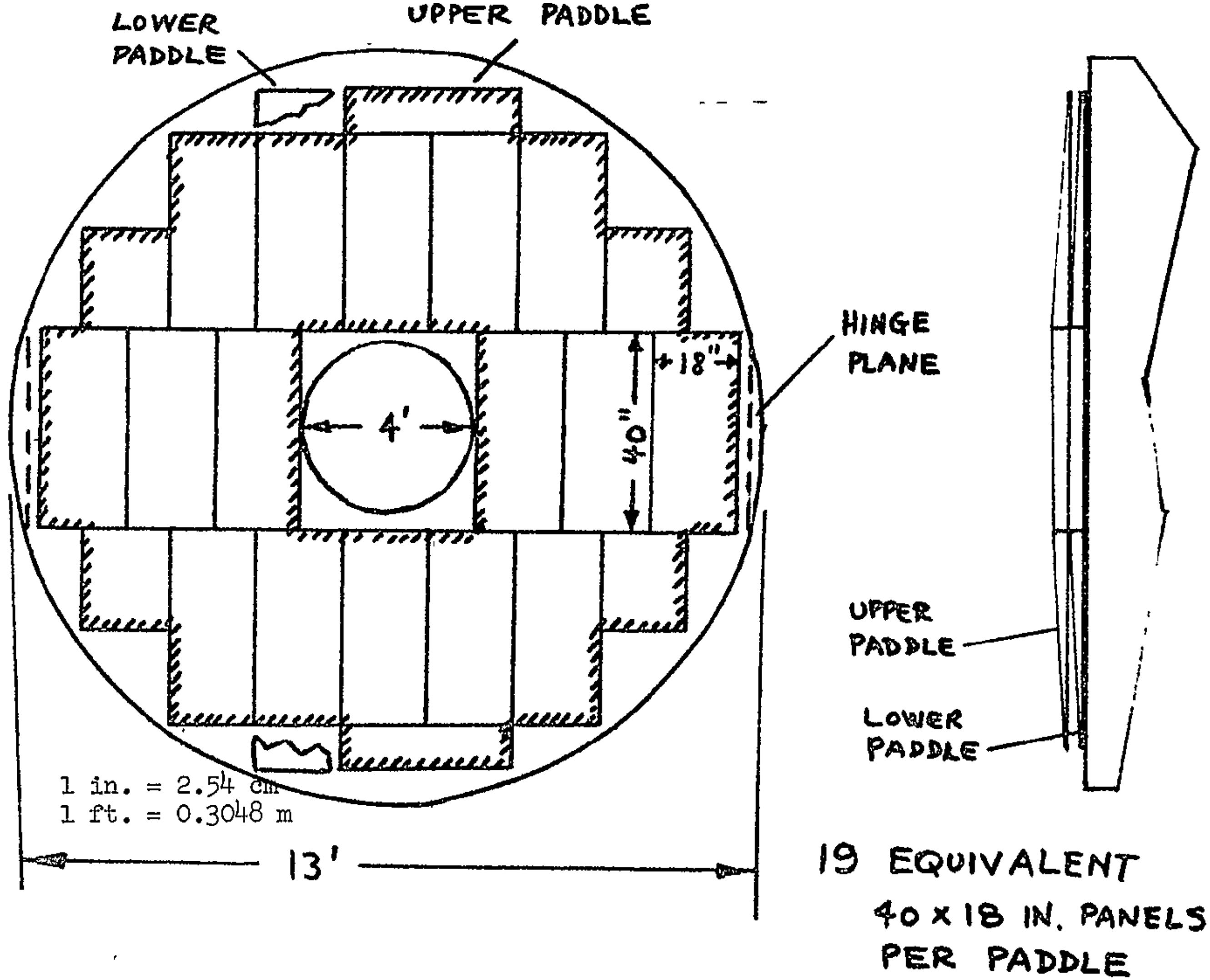


Fig. 5-19 Low-Cost OAO Solar Paddle

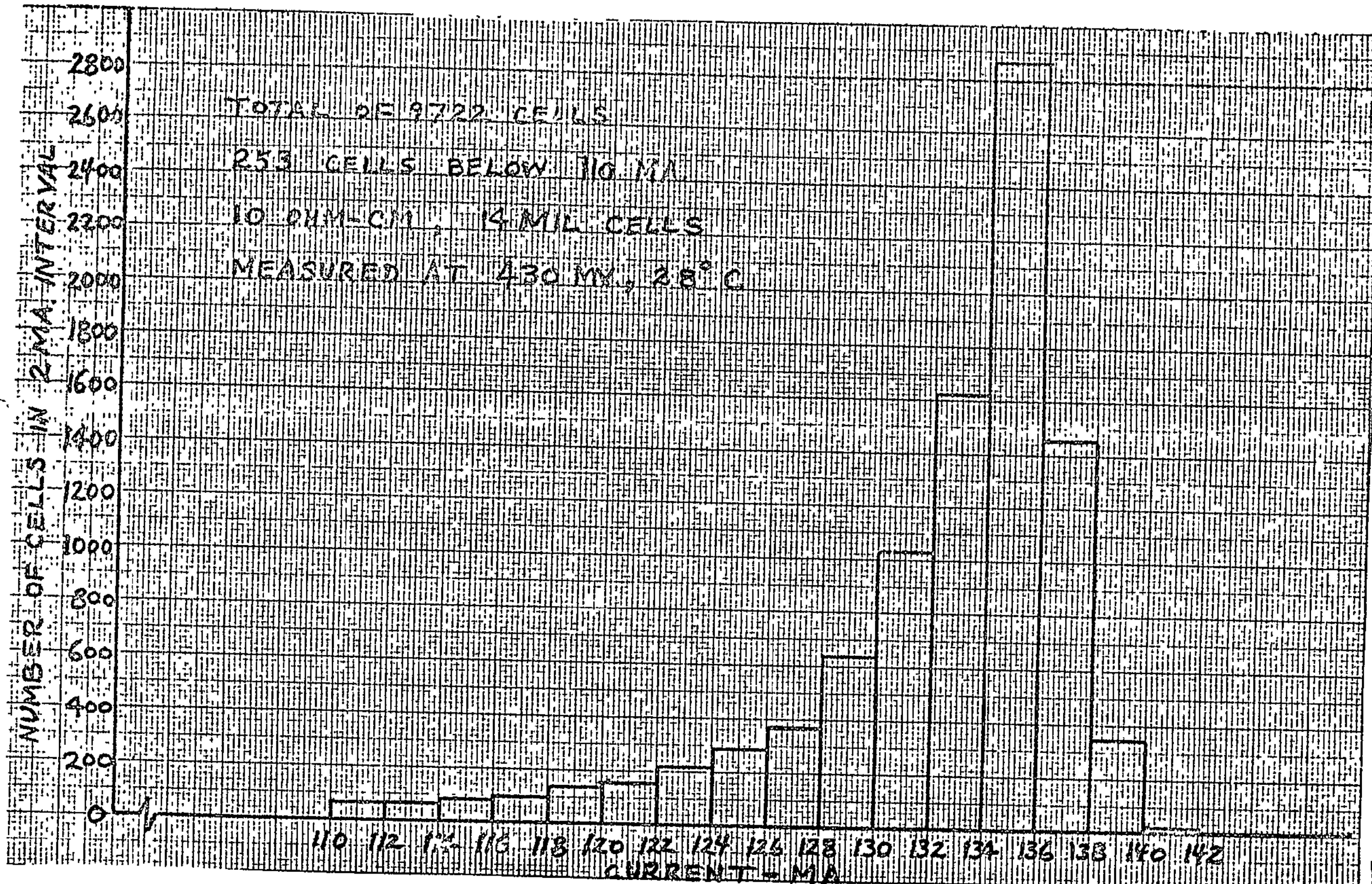


Fig. 5-20 Solar Cell Performance Distribution

When 97.5 percent of the functional cells of a lot, rather than 66.7 percent, are used, the average cost per watt of output power is reduced 30 percent and the total number of cells in an array increases 2 percent (2 percent weight increase).

Other cost saving techniques used in the design of the low-cost OAO Electrical Power subsystem include simplified, low density packaging of charge controllers, regulators, inverters and other components.

5.2.4.7 Attitude Control Subsystem (ACS). The low-cost Attitude Control subsystem of the low-cost OAO is designed to replace the following systems of the baseline OAO-B:

- Coarse Momentum Wheels (3 wheels)
- Primary Pneumatic (cold gas) System
- Secondary Pneumatic (cold gas) System
- Magnetic Wheel Unloading System

These hardware elements are described in succeeding paragraphs.

a. Baseline ACS Elements

(1) Coarse Momentum Wheels - The Coarse Momentum Wheels were used to slew the spacecraft from one pointing attitude to another. Each wheel and its drive motor were an integrated unit. The motor was built "inside-out" with the stator inside rather than outside of the rotor. Thus, the rotor has a larger moment of inertia than it would have if it were a conventional design. The motor was a 40 pole, two-phase device that could not start or continue to run unless voltage was present in both phases. The coarse wheels operated intermittently.

(2) Pneumatic System - The pneumatic system performed the functions of providing reaction mass and thrust for initial stabilization of the OAO, for torque unloading of the OAO and for RAPS (Rate and Position Sensor) control of the OAO. It was composed of a primary system and a secondary system which were interconnected such that both systems were "filled" through a common fill fitting and could be "dumped" through a common dump fitting. Both the primary and secondary high thrust jets were normally used for initial stabilization or restabilization. The primary low thrust jets were used for RAPS sunbathing or attitude-hold modes of operation.

The primary pneumatic system (Fig. 5-21) used pressurized dry nitrogen propellant stored in four spherical tanks, each capable of holding 8 lbs (3.6 kg) of dry nitrogen at a pressure of 3250 psi (22.4×10^6 N/m^2). Six combination solenoid valve/jet nozzle devices terme Primary High Thrust Jets were so located that there were two jets pointing in opposite directions along each of the vehicle's control axes.

The Secondary Pneumatic System, was almost identical to the Primary System. There were four tanks holding 8 lbs (3.6 kg) of dry nitrogen each at 3250 psi (22.4×10^6 N/m^2).

Six secondary High Thrust Jets were controlled in parallel with the Primary High Thrust Jets. Normally during initial stabilization and roll search, both Primary and Secondary systems were operating. Either the primary or secondary system could be shut off by ground command. This was done if small corrections in vehicle rates or orientation were desired or if the jets on one system were stuck open. The secondary high thrust jets, only, were fired by signals from the RAPS controller.

The eighteen nozzle assemblies used in both the Primary and Secondary Pneumatic systems are similar. All assemblies utilize the same

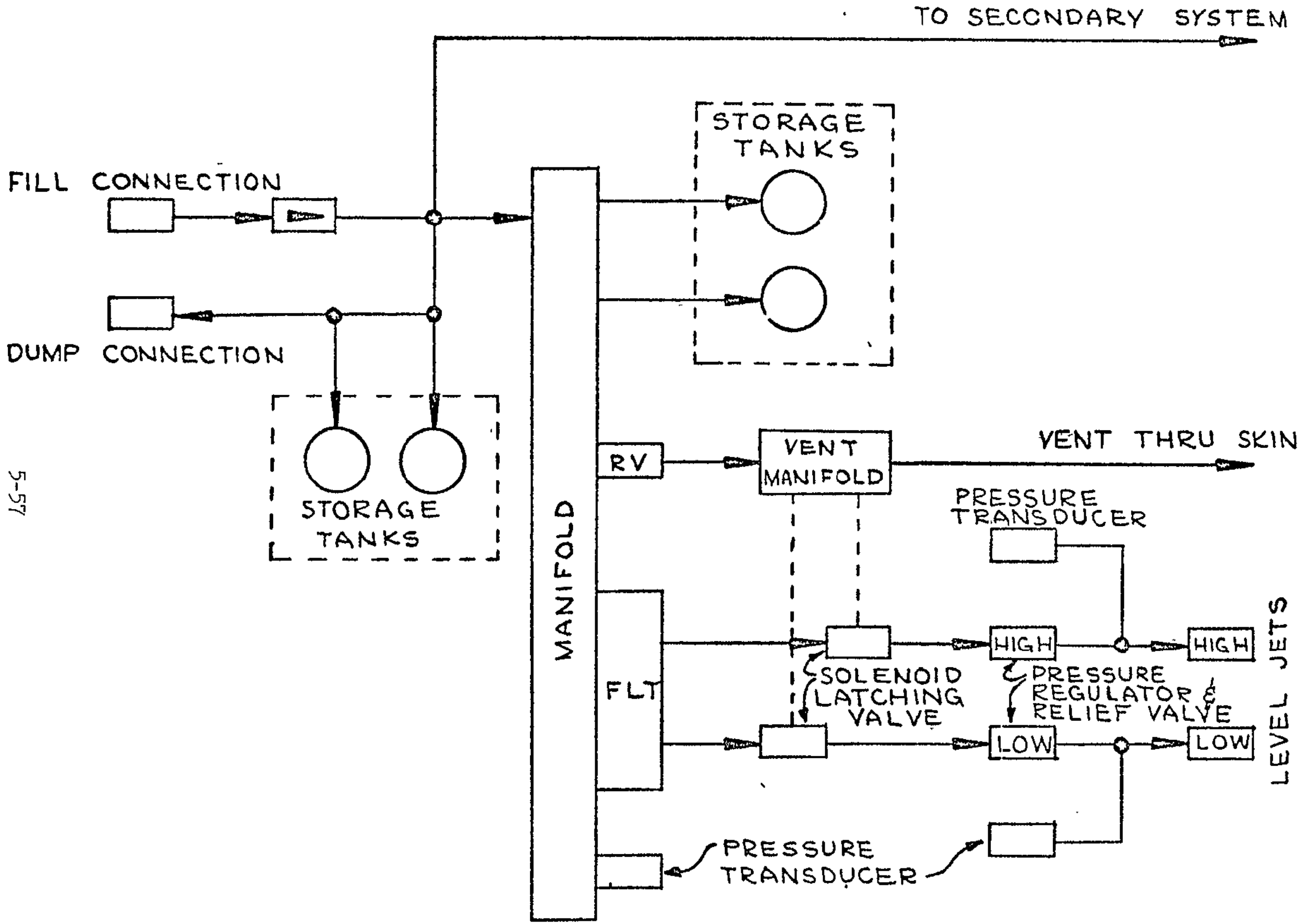


Fig. 5-21 Primary Pneumatic System

type of solenoid shutoff valve mechanism. The jet nozzle ends differ with some set at an angle other than 90° to the structural surface to which they are secured. In space, thrust per nozzle at nominal pressures is 0.1 lbs (0.44 N) for the High Thrust Jets and .0021 lbs (0.009 N) for the Low Thrust Jets.

(3) Magnetic Unloading System - The Magnetic Unloading System (MUS) was a device on the OAO that provided continuous unloading of extraneous torque on the spacecraft by the generation of a magnetic field which interacted with Earth's magnetic field in such a manner as to produce a torque in the opposite sense of the disturbing torque. The MUS could have been a backup for the Low Thrust Jets, where the MUS could develop sufficient torque to counteract the extraneous torques; however, it was considered as the preferable system since its reaction was smoother and its use obviated the need to fire the Low Thrust Jets which results in a net saving in reaction mass. It was also used to unload the continuously-running fine momentum wheels, part of the Stabilization & Control subsystem.

The MUS consisted of a Magnetometer, and its electronics, which sensed the components of the Earth's magnetic field along each of the vehicle axes, a Signal Processor which performed an algebraic summation of the Magnetometer's outputs and the signals from the FWJC's, and three Torquer Bars with their associated electronics. One bar was parallel to each vehicle axis. The Torquer Bars generated the magnetic fields that interact with the Earth's field and produced a torque on the spacecraft.

b. Low-Cost OAO Attitude Control Subsystem (ACS)

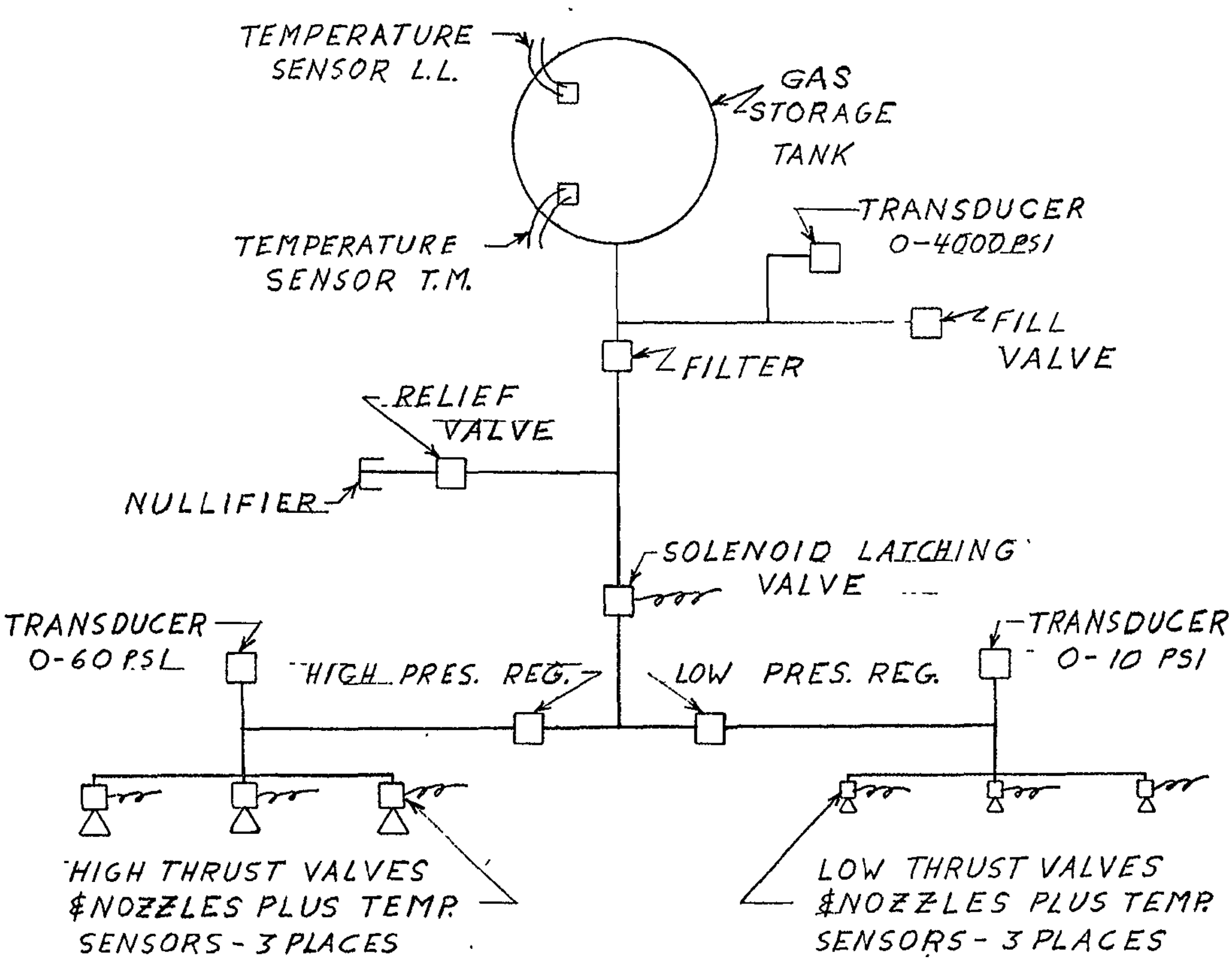
The low-cost OAO Attitude Control subsystem is a conventional cold gas control system that replaces the Coarse Momentum Wheels, the Pneumatic System and the

Magnetic Unloading system of the baseline OAO-B. It provides thrust for stabilization and slewing of the spacecraft, for unloading the fine momentum wheels, and for fine pointing of the spacecraft if the fine momentum wheels fail.

It consists of four identical modules that are installed 90° apart in the equipment section of the OAO. Each module contains three high level thrusters and three low level thrusters. The high level thrusters are used for stabilization and slewing of the vehicle. The remaining functions are performed by the low level thrusters. The schematic diagram of the module is presented in Fig. 5-22.

The gas storage tank is capable of holding 100 lbs (45.4 kg) of dry Freon 14 at a pressure of 3000 psi (20.7×10^{6} N/m^{2}), providing a total impulse of 4500 lb-sec (20,000 Ns). The tank is pressurized through a fill valve connected to the tank. The fill valve contains a needle valve which is opened to fill or bleed the tank and closed to maintain pressure in the tank. A 5 micron filter is connected into the outlet side of the Tee fitting leading to the solenoid latching valve. Also connected into line between the tank and the solenoid latching valve is the relief valve. This valve opens and vents gas when pressure rises above 3400 psi (27.6×10^{6} N/m^{2}) for any reason. The solenoid latching valve is normally open and is closed when there is a leak in the downstream system, or when there is an indication that either regulator is malfunctioning and causing an abnormal pressure to be supplied to the thrusters. The next component in the system is the high pressure regulator which reduces the tank pressure to 35 psi (0.24×10^{6} N/m^{2}) for the three high level thrusters. Each thruster produces 0.1 lb (0.44 N) thrust.

Also downstream of the solenoid latching valve the line branches to the low pressure regulator which reduces the tank pressure to 5 psi (0.034×10^{6} N/m^{2}) for the three low level thrusters. Each low-level thruster produces 0.0021 lb (0.009 N) thrust.



1 psi = 6895 N/m^2

Fig. 5-22 Low-Cost OAO Attitude Control Subsystem Module

The low-cost OAO Attitude Control subsystem module is designed to be replaced in orbit by a Space Shuttle crewman and hence emphasis is on safety. All components such as valves and regulators are operated at less than one-third of proof pressure; and the gas storage tank is designed to boiler code standards. The tank is made of aluminum rather than titanium to reduce its cost. The resulting weight penalty is acceptable to the low-cost OAO.

The cost savings resulting from the elimination of the coarse momentum wheels and the magnetic unloading system are included in the cost savings accomplished by the redesign of the OAO Stabilization and Control subsystem. Additional detail concerning the low-cost OAO Attitude Control subsystem is contained in LMSC Engineering Memo PE-5.

5.2.4.8 Environmental Control Subsystem.

a. Baseline OAO Environmental Control

The following description of the baseline OAO-B Thermal Control system is quoted from NASA SP-133, "Scientific Satellites", p. 366. The OAO structure is basically a tube, 122 centimeters in diameter and 300 centimeters long. It is surrounded by 48 truncated equipment bays, arranged in an octagonal pattern. The sequestering of OAO equipment into small bays insulated from the main satellite structure is the key to successful passive thermal control. Superinsulation made of aluminized Mylar covers all but one side of each bay (see Fig. 5-23). The bulk of the heat flowing out of (or into) each bay follows the path between the honeycomb mount and the aluminum satellite skin. Thus, the bay skins can be painted or finished in a manner appropriate to the enclosed equipment. The small equipment packages in the bays are each handled separately, as if they were small, passively controlled satellites.

The major heat input to the main OAO structure and the contained telescope is leakage through the superinsulation of the equipment bays. By careful insulation and design, heat inputs to the structure through fittings and supports are minimized. Heat leaves the structure through radiation escaping via the

open tube ends and heat transfer to nonequipment bays and end skin sections. Since the heat sinks are difficult to control, the heat flow into the structure is varied by changing the amount of superinsulation around the bay walls. The design temperature of the cylinder and contained telescope in the OAO is about $-30^{\circ} \pm 15^{\circ}C$ ($243^{\circ} \pm 15^{\circ}K$).

The most significant statement in the above description of the OAO-B Thermal Control system is the following:

"The small equipment packages in the bays are each handled separately, as if they were small, passively controlled satellites." Such a design requires a very extensive analytical effort and proliferation of detail design data. In addition, the testing required for verification of the design is extensive and complex.

b. Low-Cost OAO Thermal Control System

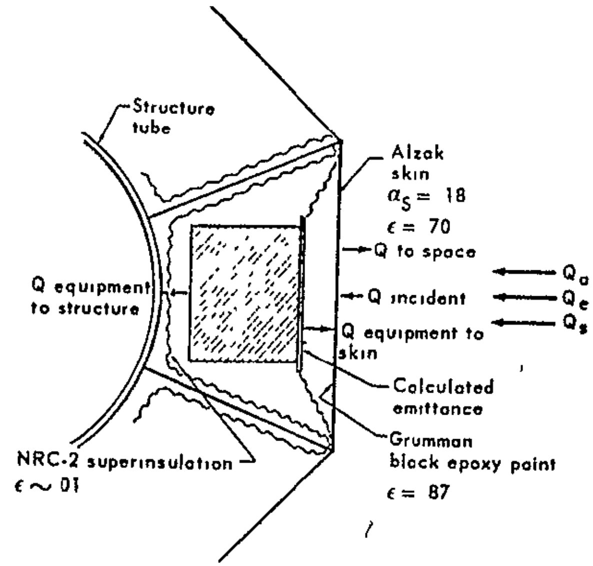
The low-cost OAO Thermal Control system consists only of passive thermal control materials as shown in Fig. 5-24. The thermal isolation of the Experiment Packag from the Equipment Section, made possible by the elimination of weight and volum constraints, results in significant simplification of the thermal control of the Low Cost OAO relative to that of the baseline OAO-B.

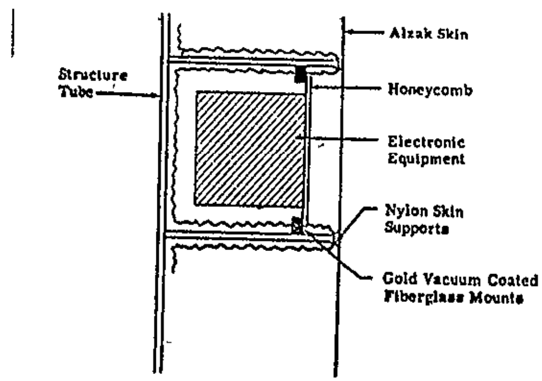
When the Experiment Package is isolated from the Equipment Section, the temperature of that section may be permitted to vary over a relatively wide range, thermal isolation and control of individual equipment units is eliminated or greatly reduced, and significant cost savings are realized.

Additional detail concerning the low-cost OAO Environmental Control subsystem is contained in LMSC Engineering Memo, PE-6.

5.2.4.9 Summary Weights of Low-Cost OAO. The weight summary, by subsystem, for the low-cost OAO is shown in Fig. 5-25.

Plan view of an OAO equipment bay, showing an instrument package surrounded by super-insulation on all sides except the one facing the outer skin.





Side view of the OAO equipment bay

Fig. 5-23 Thermal Control of OAO-B Equipment Bays

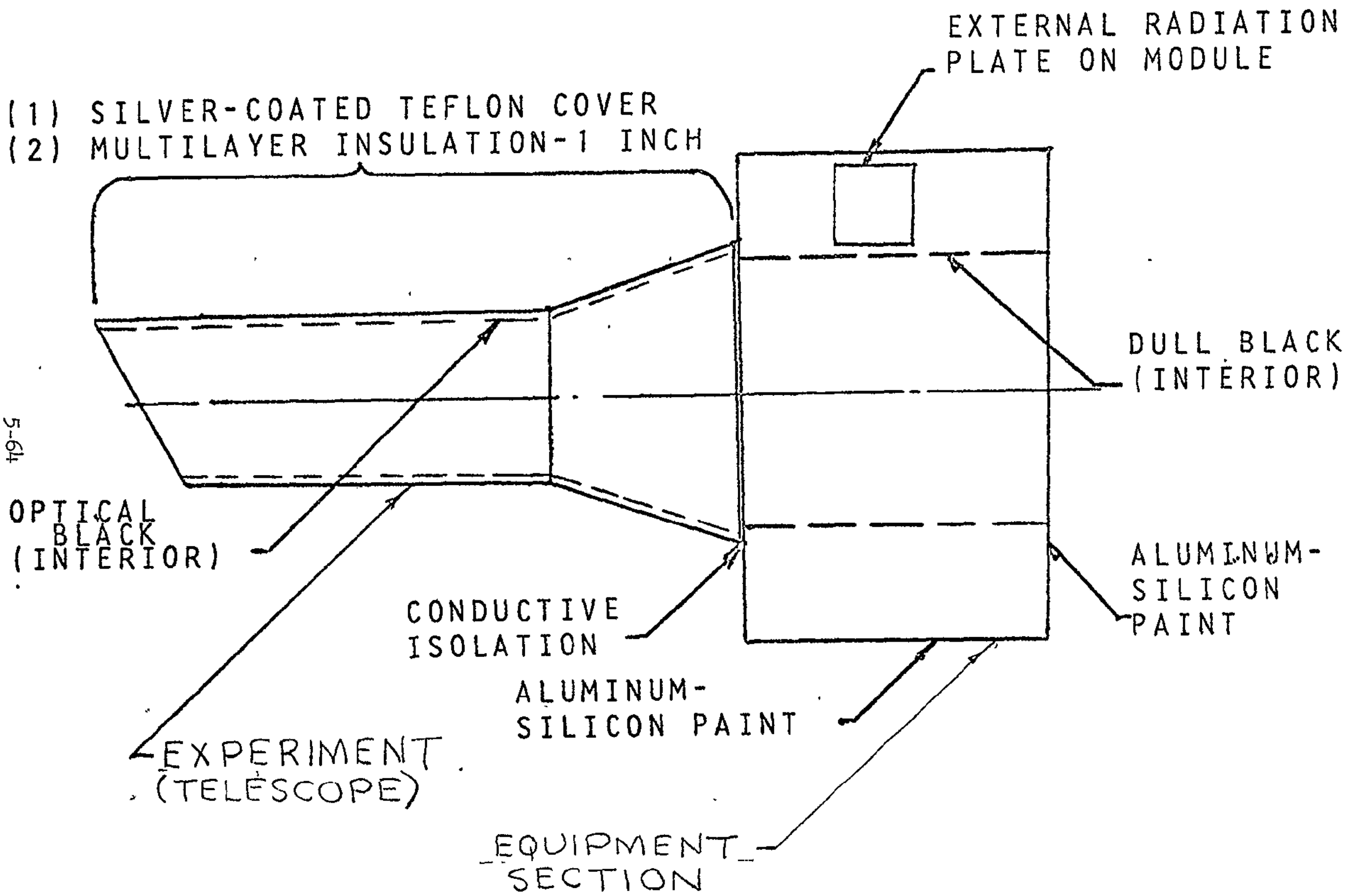


Fig. 5-24 Low-Cost OAO Thermal Control (Passive)

LMSC-A990556

HARDWARE ELEMENT	CONTINGENCY	WEIGHT
EXPERIMENT PACKAGE	15%	1970 LB
STRUCTURE & MECHANISMS	10%	1762
ELECTRICAL POWER	20%	1775
ATTITUDE CONTROL	15%	883
STABILIZATION & CONTROL	15%	655
COMMUNICATIONS, DATA PROCESSING, & INSTRUMENTATION	15%	443
ENVIRONMENTAL CONTROL	15%	100
	TOTAL DRY WEIGHT	7588 LB
ATTITUDE CONTROL GAS (FREON 14)		320
	TOTAL PAYLOAD WEIGHT	7908 LB

1 lb = 0.4536 kg

Fig. 5-25 Weight Summary - Low-Cost OAO (Shuttle-Launched)

5.2.4.10 Reliability. Because the ground rule of the study was to design a low-cost OAO with equivalent performance to the baseline OAO, very little effort has been expended toward further refinement of the baseline OAO subsystem-level reliabilities.

A summary product reliability has been calculated for the low-cost subsystems, using (1) specific redundancies in component arrangement and (2) specific internal part redundancy within components. Only two subsystems, the S&C and CDPI, have changes in reliability relevant to the baseline. Figure 5-26 summarizes the reliability data by subsystem.

5.2.5 Expendable-Launched OAO Performance and Design Requirements

The low-cost expendable-launched OAO has been designed in accordance with the requirements of LMSC-A987890-A, "General Specification - Performance and Design Requirements for Low-Cost Orbiting Astronomical Observatory", dtd 5 May 1971. It is to be launched by either of the following expendable launch vehicles:

- Atlas/Centaur
- Titan III/L2

The mission of the expendable-launched OAO is the same as that of the Shuttle-launched OAO.

The expendable-launched OAO will be mated with the upper stage of either of the two expendable launch vehicles and will be protected during ascent by a jettisonable exit fairing. The launch vehicle will inject it into the mission orbit.

5.2.6 Low-Cost Expendable-Launched OAO Configuration

The configuration of the expendable-launched OAO is identical to that shown for the Shuttle-launched OAO with two exceptions: (1) Because the length of payloa

SUBSYSTEM	BASELINE OAO	LOW-COST OAO
EXPERIMENT	.940	.940
STRUCTURE & MECHANISMS	.998	.998
ATTITUDE CONTROL	.998	.998
S&C (GNSC)	.842	.840
CDPI (TT&C)	.868	.870
ELECTRICAL	.890	.890
ENVIRONMENTAL CONTROL	.999	.999
PRODUCT	.609	.609

Fig. 5-26 Comparative Reliabilities for OAO

is not as critical with the expendable launch system (a dedicated mission with a single OAO), the sun shield is not retractable. (2) The aft end of the Equipment Section will include a short ring section which is the forward half of the LMSC Super Zip system for separation of the OAO from the expendable launch vehicle. The two solar paddles are folded flat against the aft bulkhead of the Equipment Section for launch/ascent.

5.2.7 Description of Low-Cost Expendable-Launched OAO

5.2.7.1 Subsystems of the OAO. The designations of the subsystems of the expendable-launched OAO are the same as those of the Shuttle-launched OAO. Also, the descriptions of the Shuttle-launched OAO subsystems are, in general, applicable to the expendable-launched OAO subsystems and will not be repeated. Only the differences between the subsystems of the expendable-launched OAO and the Shuttle-launched OAO will be described. The differences are summarized in Fig. 5-27.

5.2.7.2 Structures and Mechanisms Subsystem. The only change to the structure is the addition of a flanged ring to the aft end for attachment of an LMSC Super Zip separation system.

The structure and mechanisms of both the expendable-launched and the Shuttle-launched OAO are designed with high factors of safety for expendable launch vehicle loads to minimize design, analysis, manufacturing, and testing costs. The weight penalty incurred is acceptable for both the expendable-launched and Shuttle-launched OAOs.

5.2.7.3 Experiment Package. In the design of the low-cost Shuttle-launched OAO Experiment Package, some redundant equipment was deleted from the baseline OAO-B Experiment Package design. The equipment deleted was as follows:

1	+6 VDC Power Supply
1	-28 VDC Power Supply
1	Fine Guidance Detector Assembly
1	Fine Guidance Channel Selector

PAYLOAD INTEGRATION HARDWARE

- ADAPTER - PAYLOAD TO CENTAUR (ACE)
- ADAPTER - PAYLOAD TO TIII-L2 (LCE)
- EXIT FAIRING - CENTAUR "LONG" (ACE)
- EXIT FAIRING - TIII-L2 STANDARD

EXPERIMENT SUBSYSTEM

- + 6V DC POWER SUPPLY
- - 28V DC POWER SUPPLY

} IN ELECTRONIC MODULE

- DETECTOR ASSEMBLY
- FINE GUID. CHANNEL SELECTOR

} IN ELECTRO-MECH. MODULE

ELECTRICAL POWER SUBSYSTEM

- POWER CONTROLLER UNIT
- POWER REGULATOR

COMMUNICATIONS, DATA PROCESSING, INSTRUM. SUBSYSTEM

- NARROW-BAND TRANSMITTER
- WIDE-BAND TRANSMITTER
- EMERGENCY COMMAND DECODER
- ADD'L. COMPUTER SOFTWARE (25%)

STABILIZATION & CONTROL SUBSYSTEM

- INERTIAL REFERENCE UNIT
- FINE SOLAR ASPECT SENSOR
- FSAS ELECTRONICS
- GIMBALLED STAR TRACKER
- GST ELECTRONICS

Fig. 5-27 Equipment Added to Low-Cost OAO for Expendable-Booster Launch

It had been added to the Experiment Package of the baseline OAO-B to increase the probability of mission success, but was determined not to be necessary for the Experiment Package of the Shuttle-launched OAO; because in-orbit checkout prior to deployment of the OAO provided the required confidence of mission success.

However, the low-cost expendable-launched OAO, just like the baseline OAO-B, is irrevocably committed at liftoff of the launch vehicle; and the added confidence afforded by the redundant equipment is essential to the making of the decision to launch.

5.2.7.4 Stabilization and Control Subsystem. The following redundant equipment was restored to the Stabilization and Control subsystem for the same reasons presented in paragraph 5.2.7.3:

1 Inertial Reference Unit
1 Fine Solar Aspect Sensor
1 Fine Solar Aspect Sensor Electronics
1 Gimballed Star Tracker
1 Gimballed Star Tracker Electronics

5.2.7.5 Communication, Data Processing, and Instrumentation Subsystem. The following redundant equipment was restored to the CDPI subsystem for the same reasons presented in paragraph 5.2.7.3:

1 Narrow-Band Transmitter
1 Wide-Band Transmitter
1 Emergency Command Decoder

5.2.7.6 Electrical Power Subsystem. The following redundant equipment was restored to the Electrical Power subsystem for the same reasons presented in paragraph 5.2.7.3:

1 Power Controller Unit

1 Power Regulator

5.2.7.7 Other Subsystems. For the other subsystems of the expendable-launched low-cost OAO, the design is identical to the Shuttle-launched version.

5.2.7.8 Summary Weights of the Low-Cost OAO. The weight summary by subsystem, for the low-cost expendable-launched OAO is shown on Fig. 5-28.

HARDWARE ELEMENT	CONTINGENCY	WEIGHT
EXPERIMENT PACKAGE	15%	1985 LB
STRUCTURE & MECHANISMS	10%	1787
ELECTRICAL POWER	20%	1859
ATTITUDE CONTROL	15%	883
STABILIZATION & CONTROL	15%	726
COMMUNICATION, DATA PROCESSING, & INSTRUMENTATION	15%	457
ENVIRONMENTAL CONTROL	15%	100
	TOTAL DRY WEIGHT	7797 LB
ATTITUDE CONTROL GAS (FREON 14)		320
	TOTAL PAYLOAD WEIGHT	8117 LB
PAYLOAD INTEGRATION HARDWARE (ADAPTER)		291
1 lb = 0.4536 kg	TOTAL LAUNCH WEIGHT (LESS EXIT FAIRING)	8408 LB

Fig. 5-28 Weight Summary - Low-Cost OAO (Expendable Booster-Launched)

5.3 LOW-COST SYNCHRONOUS EQUATORIAL ORBITER (SEO) DESIGN

The designs of a low-cost Synchronous Equatorial Satellite (SEO), for launch by the Space Shuttle and by expendable launch vehicles are derived from an extrapolation of the Lunar Orbiter spacecraft and are described in the following paragraphs.

5.3.1 Derivation of the Baseline SEO

The Lunar Orbiter photographed the surface of the moon, processed the photographs on-board, converted the photographic data to a video signal by means of a scanner and transmitted the video data to earth for reconstruction of the photographs. Because its mission was to obtain photographic data, and because design and cost data were available, the Lunar Orbiter spacecraft was used to derive a baseline configuration for an SEO to perform an Earth Resources mission and to be used as the basis of comparison for the design and costs of the low-cost SEO.

To make the baseline SEO most representative of the synchronous equatorial spacecraft in the NASA traffic model, its mission life was extended to two years. Lunar Orbiter components were analyzed for compatibility with the two-year life requirement and necessary component redundancies were established by duplicating some of the Lunar Orbiter equipment. Some components were also eliminated; such as the bipropellant propulsion system used for midcourse and retro maneuvers in the Lunar Orbiter (but not required for the synchronous equatorial mission). The parts list for the baseline 2-year SEO is shown in Figs. 5-29a through 5-29c.

The flight configuration of the baseline SEO is shown in Fig. 5-30. The two solar array paddles are folded against the sides of the payload in the launch-stowed mode. The high-gain and low-gain antennas are also folded and stowed against the payload body (at 90° from the solar paddles). The diameter of the lower equipment mounting deck is approximately 5 ft (1.524 M). The structure is essentially the same as that of the Lunar Orbiter.

	QTY.	TOTAL WEIGHT (LB)*	
PAYLOAD TOTAL (INERT ONLY)	1		(1090.1)
EXPERIMENT SUBSYSTEM	1		(281.5)
Photographic Capsule	1		228.5
Camera System:	1	18.5	
9" Focal Length Lens	1		
Focal Plane Shutter	1		
Film Clamp & Advance	1		
Processor-Dryer System:	1	53.0	
Bimat Supply (incl. 21 lb Bimat)	1		
Bimat Takeup (with Torque Motor)	1		
Processor-Dryer Assy. (incl. Motor, Heater)	1		
Optical-Mechanical Scanner:	2	65.0	
Scanner	2		
Photomultiplier Power Supply	2		
Photomultiplier	2		
Video Amplifier & Ref. Voltage Generator	2		
Sweep & Sync. Electronics	2		
DC/DC Converter	2		
Line Scan Tube	2		
Readout Control Electronics	2		
High Voltage Power Supply	2		
Command/Control Programmer	1	7.0	
Structure and Support:	1	24.0	
Structure (15 lb)	1		
N_2 Cold Gas Supply (excluding N_2)	1		
Heater and Heater Controller	1		
Film Handling Assy (including film):	1	61.0	
Film Supply	1		
Supply Looper	1		
Camera Looper	1		
Takeup Looper	1		
Film Takeup	1		
Motor	1		
Film (24 lb)			
Secondary Experiments	1		53.0
Radiation Dosage Measurement System	1	2.0	
Micrometeoroid Detectors	20	5.0	
Advanced Vidicon Camera	2	46.0	

Fig. 5-29a Baseline 2-Year SEO Parts List (1 of 3)

STRUCTURES & MECHANISM SUBSYSTEM	1		(133.1)
Structure Assy.	1		101.7
Equipment Mounting Rack	1	29.5	
Upper Deck	1	23.5	
Upper Equipment Deck	1	5.5	
Support Structure	1	12.0	
Truss Structure	1	31.2	
Mechanisms			31.4
Low Gain Antenna Deployment Mechanism	1	4.0	
Hi-Gain Antenna Deployment Mechanism	1	4.0	
Solar Array Deployment and Rotation Mech.	1	4.0	
Separation Devices	1	3.0	
Solar Array Storage Devices	1	16.4	
ENVIRONMENTAL CONTROL SUBSYSTEM	1		(11.0)
Equipment Thermal Barrier	1		5.5
Camera Thermal Door	1		2.5
Tank Heaters	4		1.5
EMD Heat Exchanger	1		0.5
Thermal Coatings	1		1.0
ATTITUDE CONTROL SUBSYSTEM	1		70.3
N_2 Pressure Regulator	2	1.9	
N_2 Tank (each contains 15 lb N_2)	4	50.0	
N_2 Squib Valve	4	3.2	
N_2 Filter	2	0.8	
Test/Fill & Dump Valve	4	0.9	
Thrusters - Roll & Pitch	6	4.0	
Thrusters - Yaw	2	0.5	
Thrusters - E/W	4	1.0	
Tubing & Fittings	1	8.0	

Fig. 5-29b Baseline 2-Year SEO Parts List (2 of 3)

ELECTRICAL POWER SUBSYSTEM	1		311.7
Solar Arrays	2	80.0	
Batteries	4	120.0	
Charge Controller	4	16.0	
Shunt Regulator & Relays	4	18.4	
Other Electrical Components	1	3.3	
Cables & Connectors	1	50.0	
Array Tracking Devices (Drive Motors, Gear Box, Control, Power Take-off)	2	24.0	
COMMUNICATION, DATA PROCESSING, INSTRUMENTATION SUBSYSTEM	1		146.5
Hi Gain Antenna & Boom	1	2.3	
Modulation Selector	2	8.0	
Command Decoder	2	18.0	
Transponder	2	25.5	
TWTA	2	11.0	
Wide Band Tape Recorder and A/D Converter	2	40.0	
Low Gain Antenna	1	1.5	
PCM Multiplexer/Encoder	4	32.0	
Signal Conditioning Unit	1	3.5	
Transducers	1 set	4.7	
STABILIZATION & CONTROL SUBSYSTEM			136.0
Flight Control Electronics Unit	2	35.0	
Solar Aspect Sensor	2	6.0	
Flight Electronics Switching Unit	1	8.0	
Earth Horizon Sensor	2	20.0	
Polaris Tracker	2	30.0	
Fixed Reaction Wheels	3	27.0	
Rate Gyro Package	1	10.0	
EXPENDABLES			72.0
N_2 Camera Capsule		12.0	
N_2 Attitude Control		60.0	

Note: 1 lb = 0.4536 kg

Fig. 5-29c Baseline 2-Year SEO Parts List (3 of 3)

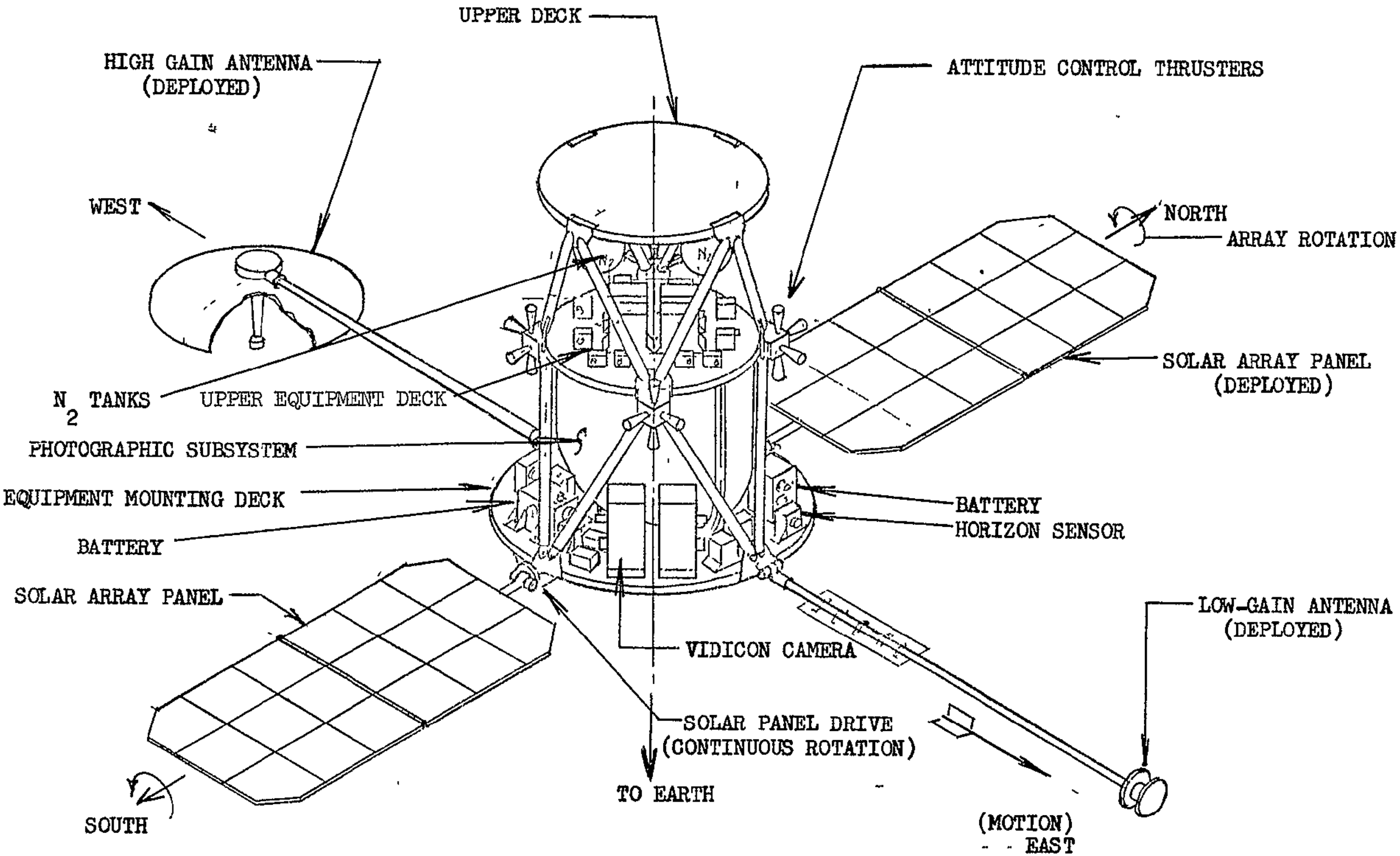


Fig. 5-30 Baseline SEO Configuration

LMSC-A990556

5.3.2 Shuttle-Launched SEO Performance and Design Requirements

The design effort on the low-cost Shuttle-launched SEO was initiated with preparation of LMSC-A981600-A, "General Specification - Performance and Design Requirements for Low-Cost Synchronous Equatorial Orbiter (Earth Resources Satellite)", dtd 5 May 1971 (Revised). The requirements therein were the same as for the Lunar Orbiter, tailored as required for the SEO mission.

The SEO is an unmanned satellite designed to gather scientific data concerning earth resources from synchronous equatorial earth orbit. Photographic data will be obtained by means of a photographic subsystem consisting of a frame type camera, a processor-dryer, and a scanner. Additionally, television data will be obtained by means of a vidicon camera. Both the photographic data and the television data will be transmitted to ground receiving stations by means of a communications link in a time-sharing mode.

The experiment packages must function continuously (in accordance with the required duty cycle) for a period of two years. Also, the spacecraft must provide controlled environment, electrical power, attitude stability, functional control, and communication for an orbital life of two years.

The SEO is designed to operate in a synchronous equatorial earth orbit [altitutude 19320 nm (35780 km)]. It will be placed into orbit by the Space Shuttle and Space Tug. The Shuttle will carry the Tug and SEO to low earth orbit, and then the Tug will deliver the SEO to synchronous equatorial orbit and return to low earth orbit.

It is planned that four SEOs will be placed into position for simultaneous operation (separate Shuttle launches), equally spaced around the earth. With the lens system and resolution selected, there will be provided full equatorial coverage and approximately 2800 nm (5186 km) coverage above and below the equator.

5.3.3 Low-Cost Shuttle-Launched SEO Configuration

The flight configuration of the low-cost SEO is shown in Fig. 5-31. In the launch-stowed condition, the solar paddles are rotated into a fore-aft plane (perpendicular to the plane of the docking ring). Access doors on each of two sides of the SEO structure can be readily opened by Shuttle cargo crew to allow replacement of modules mounted on internal shelving. Access to the SEO as mounted within the Shuttle cargo bay is possible.

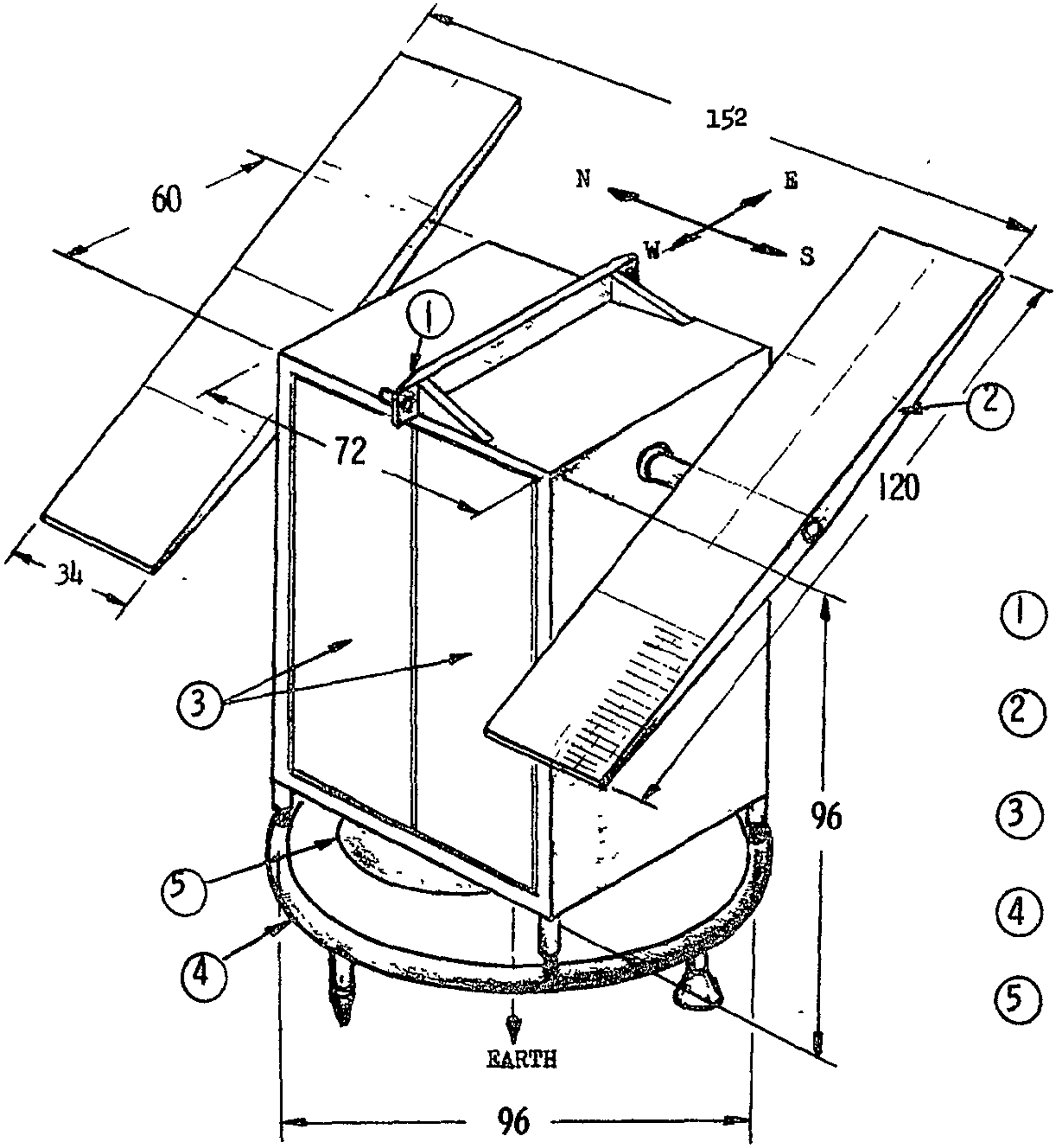
To provide for the docking of the Space Tug to the SEO, a docking ring and corner reflectors are mounted on the SEO. It has been assumed that all active docking provisions will be on the Space Tug; the SEO will provide only a passive but stable target for the rendezvous, terminal approach, and docking.

5.3.4 Description of Low-Cost Shuttle-Launched SEO

5.3.4.1 Subsystems of the SEO. The subsystems comprising the SEO, both Shuttle-launched and expendable-launched, and the LMSC Engineering Memos that describe them in detail are as follows:

Subsystem	LMSC Engineering Memo
Experiments	PE-21
Stabilization & Control	PE-22
Communications, Data Processing, and Instrumentation	PE-23
Electrical Power	PE-24
Attitude Control	PE-25
Environmental Control	PE-26
General Description of Payload	PE-27

Summary descriptions of all subsystems are presented in the following paragraphs. A separate Engineering Memo describing the Structures and Mechanisms Subsystem has not been prepared; that subsystem is described first.



① Shuttle Support Lug (IL/IR)

② Solar Arrays - Sun-Orienting

③ Access Doors to Modules (opposite side same)

④ Docking Ring Assy

⑤ Hi-gain Antenna

1 in. = 0.0254 m

Fig. 5-31 Low-Cost SEO General Configuration

5.3.4.2 Structures and Mechanisms Subsystem. The primary structure of the low-cost SEO is shown in Fig. 5-32. The structure is a rectangular box divided into compartments in which readily-removable equipment modules are installed. The modules are designed to be removed and replaced by a space-suited Shuttle crewman. The structure is a riveted assembly made up primarily of aluminum sheet and extrusions. Stainless steel angles provide thermal isolation of the top and bottom panels, and stainless steel hinges thermally isolate the doors. Factors of safety of three or greater are used to reduce design and analysis efforts and to reduce or eliminate static load testing.

Because of the need for orbit docking with Space Tug, a docking ring assembly has been provided on the SEO. The ring is designed to be compatible with space docking provisions on other orbiting vehicles such as the Space Station segments. A nominal diameter of 96 in. (2.44 m) has been chosen. Figure 5-31 shows the general configuration of the ring. Each of the four posts connecting the ring to the base of the SEO structure assembly is a telescoping tube assembly with a set of balanced internal springs. These springs are installed in the compressed condition so that a load higher than the preload must be applied before the spring will deflect further. The preload will be set to exceed the maximum g-load imparted by the Space Tug during orbit-to-orbit boost of the SEO. The function of these springs is two-fold:

(a) Deflect to allow lengthening or shortening of the Shuttle between the fixed-pin structural mountings of the SEO and the Tug (up to 2 in. (5.1 cm) in either direction).

(b) Aid in attenuating docking shock loads.

The docking ring assembly is also provided with a set of two drogue funnels @ 180° and a set of probes @ 180° and displaced 90° from the funnels; these four elements are intended to mate with similar elements provided on the Space Tug. All latching of the mated probes and funnels is planned to be accomplished mechanically (automatic). Unlatching provisions (solenoid) are assumed to be on the Tug; all docking elements on the SEO are functionally passive.

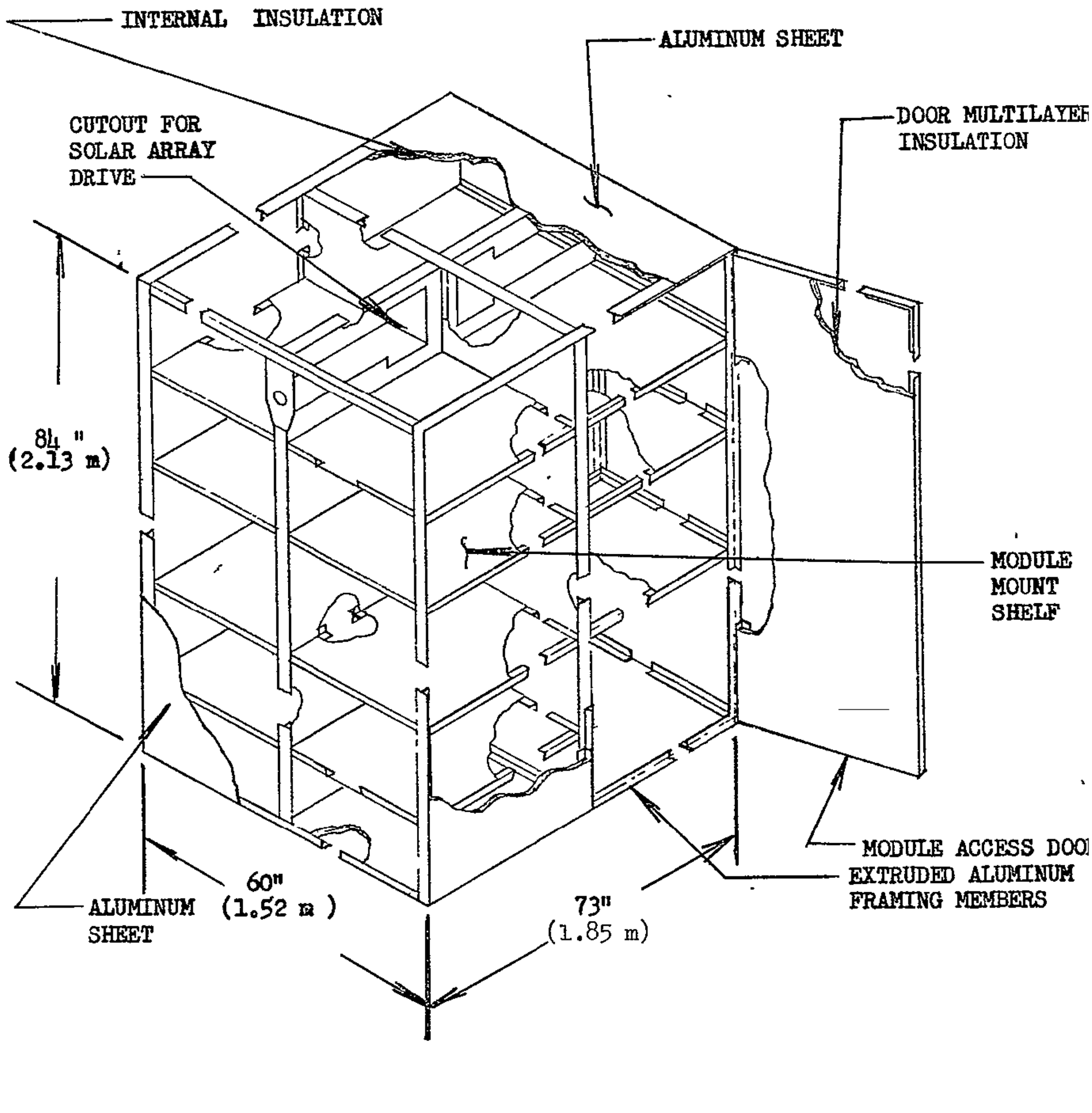


Fig. 5-32 Low-Cost SEO Equipment Section Structure

A breakdown and the weights of the primary structure and the docking ring assembly are given in Fig. 5-33.

The SEO equipment module arrangement is shown in Fig. 5-34. The hardware elements comprising each module, and the module weights, are listed in Figs. 5-35a, 5-35b, and 5-35c. The configurations of the modules have been determined as a result of requirements to have reasonable module sizes, weights, and unit costs; to provide the required overall SEO weight and balance (mass distribution); and to combine similar functional equipment into the same module. The overall dimensions of the larger modules are:

Experiment - Photographic Module No. 1	30 x 30 x 48 in.
CDPI Modules Nos. 1, 2, 3	14 x 24 x 30 in.
S&C - Modules Nos. 1, 2	14 x 24 x 30 in.

Note: 1 in. = 2.54 cm

5.3.4.3 Experiment Subsystem. The Experiment subsystem consists of three basic elements; the photographic system; the advanced vidicon camera system; and two secondary experiments, a radiation detector and a set of 20 micrometeoroid detectors. Because of the relatively low cost of the secondary experiments no effort has been expended to achieve a low-cost design equivalent. A low-cost design has been developed for the principal experiments, with primary emphasis upon the high-cost photographic system.

a. Photographic System - The design of the Eastman-Kodak photographic system for the Lunar Orbiter was based upon lunar mission requirements and rather severe weight limitations. Redefined mission requirements for a synchronous equatorial earth resources mission plus removal of constraints upon size and weight allow a larger, heavier, and simpler system with extended mission life capabilities. The basic Lunar Orbiter concepts have been utilized.

SPACECRAFT STRUCTURE ASSY.

1	Top Panel	30.1 lb
1	South Panel	49.4
1	South Vertical Divider	26.4
1	Middle Vertical Divider	47.0
1	North Vertical Divider	25.6
1	North Panel	51.4
1	Bottom (Earth Side) Panel	27.0
14	Horiz. Shelf Unit	85.4
16	Module Hold-Down Devices	96.0
4	East/West Doors	142.0
		580.3 lb
	10% Contingency	58.0
	Total	638.3 lb

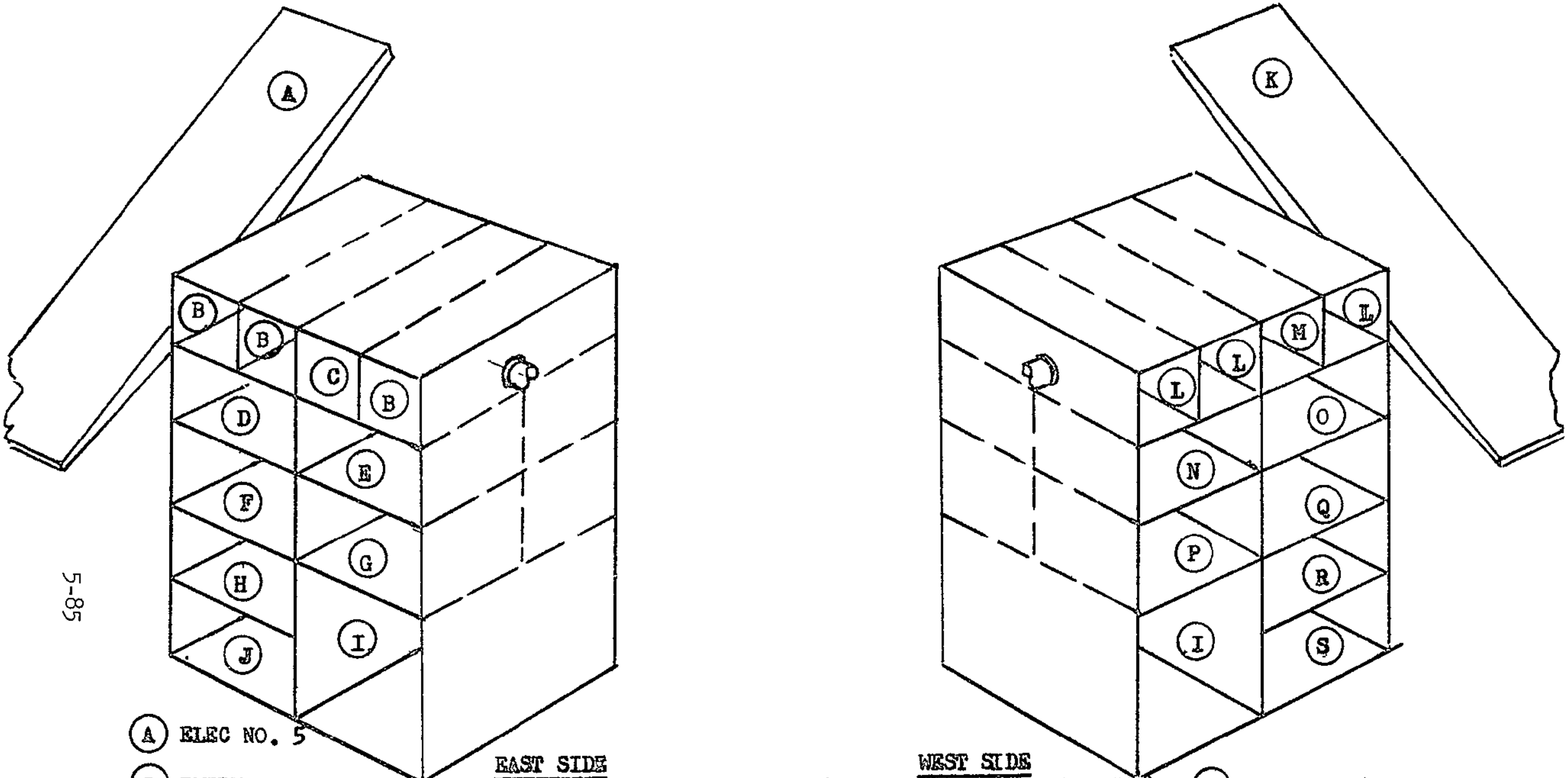
DOCKING RING ASSY.

4	Shock/Length Adjust. Device	32 lb.
1	Docking Ring - 96" O.D.	53
2	Drogue Funnels	3
2	Probe Assy.	6
		94 lb
	10% Contingency	9
	Total	104 lb

Spacecraft Structure	638 lb
Docking Ring Assy	104
Structures & Mech. Subsystem	742 lb

1 lb = 0.4536 kg

Fig. 5-33 Breakdown and Weights - Structures & Mechanisms Subsystem



A ELEC NO. 5
B EMPTY
C ELEC NO. 3
D ELEC NO. 1
E CDPI NO. 1
F ACS NO. 1
G ACS NO. 2
H CDPI NO. 3
I EXPER. NO. 1
J EXPER. NO. 2

K ELEC NO. 6
L EMPTY
M ELEC NO. 4
N CDPI NO. 2
O ELEC NO. 2
P ACS NO. 3
Q ACS NO. 4
R S&C NO. 1
S S&C NO. 2

Fig. 5-34 SEO Module Location and Arrangement

SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT
Electrical	Battery No. 1	o NiCd Battery (4) - 12 Amp. Hr. o Current Shunt Assy. o Module Base/Cover o Internal Cables	Basic 154 lb 10% Cont. 16 Total 170 lb
Electrical	Power Control No. 2	o Battery Charge Controller (4) o State-of-Charge Unit o Power Distribution Unit o Ground Power Relay o Motor Controller - S/A Drive o Pulse Generator - S/A Drive o Module Base/Cover o Internal Cables	Basic 92 lb 15% Cont. 14 Total 106 lb
Electrical	Paddle Drive (2 Req'd) No. 3 No. 4	o Drive Motor (2) o Gear Box o Module Base/Cover o Internal Cables	Basic 34 lb 15% Cont. 5 Total 39 lb
Electrical	Solar Paddle (2 Req'd) No. 5 No. 6	o Solar Cell Panel (8) o Paddle Structure o Voltage Regulator (Diodes) o Internal Wiring	Basic 60 lb 15% Cont. 9 Total 69 lb

1 lb = 0.4536 kg

Fig. 5-35a Low-Cost SEO Modules (1 of 3)

SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT
Communication, Data Processing, Instrumentation (CDPI)	Data Handling (2 Req'd) No. 1 No. 2	• Tape Recorder/Encoder • PCM Multiplexer/Encoder • Command Decoder • Module Base/Cover • Internal Cables	Basic 70 lb 15% Cont. 11 Total 81 lb
Communications, Data Processing, Instrumentation	Communications No. 3	• Transponder (2) • TWTA • Modulation Selector (2) • Module Base/Cover • Internal Cables	Basic 75 lb 15% Cont. 12 Total 87 lb
Stabilization & Control	Sensing & Flight Control Electronics No. 1	• Flight Control Electronics • Horizon Sensor (2) • Solar Aspect Sensor (2) • Module Base/Cover • Internal Cables	Basic 85 lb 15% Cont. 13 Total 98 lb
Stabilization & Control	Momentum No. 2	• Gimballed Momentum Wheel • Wheel Drive Electronics • Rate Gyro Package • Wheel Safety Shield • Module Base/Cover • Internal Cables	Basic 109 lb 15% Cont. 16 Total 125 lb

1 lb = 0.4536 kg

Fig. 5-35b Low-Cost SEO Modules (2 of 3)

SUBSYSTEM	MODULE	EQUIPMENT IN MODULE	MODULE WEIGHT
Attitude Control	Attitude Control (4 req'd) No. 1 No. 2 No. 3 No. 4	• Tank • Valves: Fill, Relief, Solenoid Latching • Press. Regulator • Thrust Valve Cluster with 4 Nozzles • Plumbing • Transducers • Internal Support • Module Base/Cover • Internal Cabling	Basic 125 lb 15% Cont. 18 Total $\boxed{143 \text{ lb}}$
Experiment	Photographic No. 1	• Camera • Processor/Dryer • Optical-Mech. Scanner (2) • Film Handling Assy. • Electronic Assy. • Nitrogen System • Film Supply • Bimat Supply • Internal Structure • Case • Internal Cabling	Basic 365 lb 15% Cont. 54 Total $\boxed{419 \text{ lb}}$
Experiment	Vidicom Camera No. 2	• Vidicon Camera Assy (2) • Module Base/Cover • Internal Cables	Basic 75 lb 15% Cont. 11 Total $\boxed{86 \text{ lb}}$

1 lb = 0.4536 kg

Fig. 5-35c Low-Cost SEO Modules (3 of 3)

The low-cost Photographic system has been designed as a readily replaceable module. The entire Photographic system is track-mounted within a sealed container and can be removed from the container readily when the container end cap is removed. The container is approximately 32 in. (0.813 m) diameter and 48 in. (1.219 m) long.

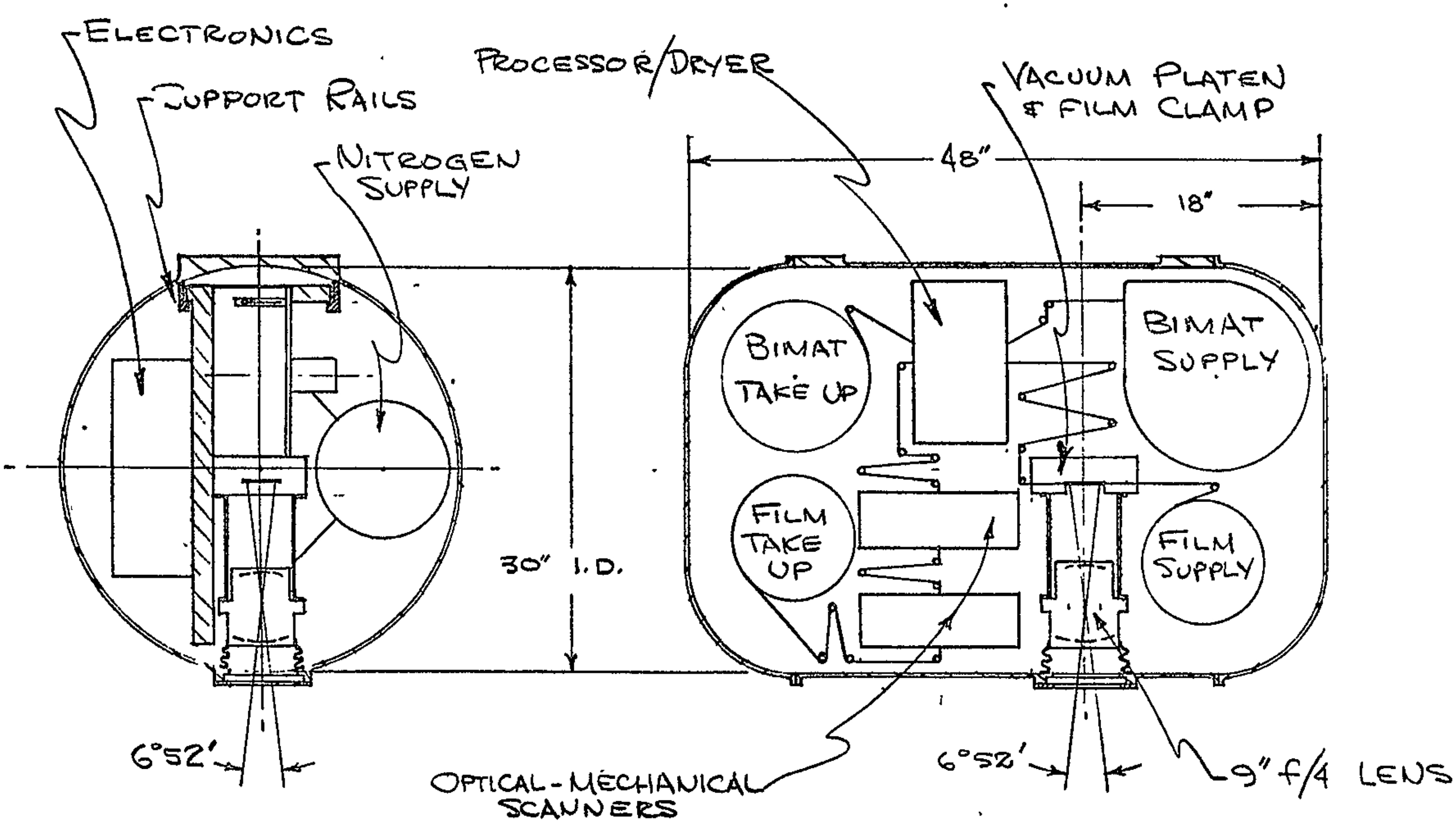
The general configuration of the Photographic system is shown in Fig. 5-36.

A single lens system is used to produce high resolution images on 70 mm film. The lens is a Bausch & Lomb 9 in. (22.9 cm) focal length f/4.0 Super Baltar lens operating at fixed aperture. A between-the-lens shutter, having speeds of 10, 20, and 40 milliseconds, selectable by ground command, is used. Also, the metal leaf, between-the-lens shutter, precludes the need for an additional sun shutter. During exposure, the film is held flat by film clamps and vacuum.

At synchronous altitude, resolution of 1.24 nm (2.40 km) will be obtained at at nadir with an object contrast ratio of 3:1.

The film to be used is Eastman-Kodak Type SO-243 which has an aerial exposure index of 1.6. This film, although relatively slow, has extremely fine grain and high resolving power. Also, it has low sensitivity to ionizing radiation. To provide for control and calibration, data consisting of high and low contrast resolution bars, a ten step gray scale, and a linearity pattern is pre-exposed along one edge of the film under precisely controlled conditions.

Eastman-Kodak "Bimat" (SO-111) processing is used to develop the exposed SO-243 film at a processing rate of 2.3 in. (5.84 cm) per minute. The exposed film is laminated to the Bimat film and processing goes to completion in 3.4 min. of contact time at a temperature of 85°F (303°K). As the film and Bimat leave the processor drum, they are delaminated, with the Bimat film going to a dryer drum which is at a temperature of 95 ± 3°F (308 ±2°K). Residual moisture on the film is collected by a mat containing potassium



1 in. = 2.54 cm

Fig. 5-36 Low-Cost Photographic System

thiocyanate desicant. The dried frame is stored on a takeup looper for readout. After each sequence of five photographs, the desicant mat is heated to drive off acquired moisture.

Bimat storage is a problem because of its limited storage life. The most suitable storage conditions are a high moisture atmosphere with temperature maintained in the 33° to $40^{\circ}F$ (274° to $278^{\circ}K$) range. The Bimat supply system will be designed to maintain these conditions. In order to facilitate the temperature control of the Bimat supply, the Photographic subsystem will be thermally isolated from the remainder of the spacecraft and will be designed to stabilize in the temperature range of 20° to $30^{\circ}F$ (266° to $272^{\circ}K$). Thermostatically controlled heaters will be used to raise the temperature of the Photographic system into the 33° to $40^{\circ}F$ (274° to $278^{\circ}K$) range and maintain it there.

The film and Bimat consumption will be six frames per day for a total of 905 ft (276 m) for a two-year mission. The film spool diameter is approximately 9 in. (22.9 cm) and the Bimat spool diameter is approximately 11 in. (27.9 cm).

The photographic images are converted to electrical video signals by the optical-mechanical scanner. The scanner consists of a line-scan tube, a movable lens which mechanically indexes the position of the line image on the film, collector optics, photomultiplier tube, and video amplifier, with the video amplifier being physically located in the electronic module. The readout time per frame is 9.5 min. Two optical-mechanical scanners, in series with one redundant, are incorporated into the Photographic system. The space between the two scanners has a takeup looper to hold each frame until ground reconstruction has been completed.

The nitrogen container will be filled with 12 lbs (5.44 kg) of nitrogen. Once every twenty-four hour period, an amount of nitrogen, sufficient to lower the relative humidity in the container to approximately 50 percent,

will be bled into the container. Nitrogen and water vapor, in excess of the amount necessary to maintain a pressure of 2.0 psia (1.38 N/cm^2) will be vented to space. In order to maintain a light tight seal between the window in the container and the lens, it is necessary to have a bellows sealed at both the lens and the window. Therefore, after the camera is installed in the container, the bellows is to be sealed to the window mounting frame and the window installed. The window, which will have one surface anti-reflection coated and the other surface coated with a minus blue filter will be installed with an "O" ring seal.

The two end caps of the container will have gasket seals capable of withstanding a differential pressure of 2 psia (1.38 N/cm^2).

The container will be thermally isolated from the spacecraft. For thermal control, the exterior surface of the center section of the container will be painted flat black and the end caps will be polished aluminum.

The entire photographic system has been reviewed with the purpose of reducing cost. The primary cost reductions were accomplished by increasing volume and weight to simplify design, by using aluminum in place of beryllium, and by the elimination of close tolerances machining for weight control.

b. <u>Advanced Vidicon Camera System (AVCS)</u>

The Advanced Vidicon Camera system is a modified version of the ESSA-AVCS System which was designed to provide earth coverage from orbital altitudes of 1400 nm (2593 km). Since the proposed Advanced Vidicon Camera System is to operate at synchronous orbit, a longer focal length lens becomes mandatory. A 46 m/m focal length-lens provides approximately the same angular field-of-view as the Photographic system. With the present vidicon, this lens has approximate ground coverage of 5500 x 5500 nm (10,186 x 10,186 km) and the resolution at the center of the field of view is approximately 5.5 nm (10.2 km).

Two systems, one standby redundant, are supplied as a bore-sighted package for each spacecraft. Fifty pictures per illuminated portion of each 24 hour period are planned. The AVCS consists of a 46 mm focal length lens with automatic iris diaphragm control ranging from f/4 to f/16; an electromagnetically controlled focal plane shutter; a 2.5 centimeter vidicon; associated sweep circuits, high voltage power supply, and preamplifier all mounted in an electronic module; and a flash lamp illuminated gray scale directly in front of the vidicon. Because of the very special nature of the AVCS and the fact it was assumed furnished GFE, no low-cost redesign was attempted.

c. Micrometeoroid Detectors

The micrometeoroid detectors which are described in detail in NASA Technical Memorandum X-810, dtd February 1963, are small pressurized cells, mounted on the exterior of the spacecraft, which when punctured by a micrometeoroid, yield an output signal change of 0 to +6 VDC. These hardware items are essentially off-shelf; no effort was devoted to low-cost design.

d. Radiation-Dosage Measurement System

The Radiation Dosage Measurement System consists of one scintillation counter, to detect charged particles and gamma radiation, and two solid state detectors which respond only to charged particles of pre-selected minimum energy levels. Here again, the low relative cost and classification of items as GFE offered no challenge to low-cost design analysis.

Additional details of the low-cost SEO Experiment Subsystem are contained in LMSC Engineering Memo, PE-21.

5.3.4.4 Stabilization and Control (S&C) Subsystem. The Baseline SEO is carried into synchronous orbit by a Burner II stage. During the transfer-orbit to synchronous orbit the Burner II/SEO combination is slowly rolling in the apogee burn attitude for gyro drift-averaging, and thermal control.

After synchronous orbit injection and stage separation, the Baseline SEO must orient itself to the earth vertical-velocity vector-orbit normal coordinate frame and maintain attitude stability on station, within specified tolerances, for two years.

The baseline SEO S&C subsystem has the following functions:

- control attitude transients from Burner II separation
- acquire Earth and Yaw references
- meet SEO pointing requirements (two degrees, all axes) for two years
- stabilize and control SEO attitude during east-west station-keeping maneuvers.
- reorient SEO to the orbit reference attitude from any attitude following loss of reference for tumbling rates up to 3 deg/sec.
- point the SEO spacecraft to the sun with near-zero rates following primary system failure.

The equipment required to mechanize the baseline SEO S&C subsystem is as follows:

- Flight Electronics Control Assembly
- Earth Horizon Sensor
- Sun Sensors (2)
- Rate Gyro Package
- Flight Electronics Switching Unit
- Polaris Star Tracker
- Reaction Wheels (3)

This equipment is derived from the equivalent Lunar Orbiter equipment retaining maximum commonality of operating and design concepts, and the constraints of weight and volume limitations.

In contrast to the baseline SEO, the low-cost SEO is carried to its final position, activated, and checked out aboard the Space Tug, and separated in its operational attitude.

If the low-cost SEO should fail to pass all critical tests before Tug departure, it can be retrieved and returned to low orbit or to earth for repair. Should the SEO fail during the two-year life it may be retrieved for reuse or salvage.

The low-cost SEO S&C subsystem has fewer functions and is simpler than that of the Baseline SEO. Figure 5-37 compares the implementation of the two SEO spacecraft S&C subsystems and Fig. 5-38 compares the operational modes for the two designs.

The low-cost SEO stabilization and control subsystem shown on Fig. 5-39 is based on active control of a two-degree-of-freedom CMG. Attitude control torques in pitch are obtained by varying the wheel speed while roll and yaw control torques result from tilting the wheel spin axis. The momentum of the gimballed wheel supplies gyroscopic restraint of yaw attitude. A horizon sensor is the source of earth-pointing error signals and scanning laser radar on the Space Tug provides pitch, yaw and roll attitude errors for docking.

Two wide-angle digital solar aspect sensors and a 3-axis rate gyro package are included to aid in reorientation to earth reference in the event of a reversible system failure (e.g., intermittent power supply outage). The same units, in conjunction with cold gas thrusters, comprise an anti-tumbling system to permit safe revisit after subsystem failure.

The Flight Programmer section of the Flight Electronics Control Assembly provides timing, sequencing, and logical computations for all SEO subsystems. It accepts commands for real-time execution and for storage from the communications (CDPI) subsystem. The Closed Loop Electronics Section converts sensor signals and command messages into control logic and actuation signals.

ITEM OF EQUIPMENT	APPLICATION	
	BASELINE SEO	LOW-COST SEO
Flight Electronics Control Assembly	Yes	Yes
Earth Horizon Sensor	Yes	Yes
Sun Sensors (2)	Yes	Yes
Rate Gyro Package	Yes	Yes
Flight Electronics Switching Unit	Yes	No
Polaris Star Tracker	Yes	No
Reaction Wheels (3)	Yes	No
Gimballed Momentum Wheel	No	Yes
Docking Reflectors	No	Yes

Fig. 5-37 Comparison of Baseline and Low-Cost SEO S&C Subsystem Implementation

OPERATING MODE	APPLICATION	
	BASELINE SEO	LOW-COST SEO
Initial Earth Reference Acquisition	Yes	No
Initial Yaw Reference Acquisition	Yes	No
Yaw Reference Hold	Yes	No
Earth Reference Hold	Yes	Yes
East-West Stationkeeping	Yes	Yes
Attitude Reference Reacquisition	Yes	Yes
Momentum Unload	Yes	Yes
Hold on Gas (Backup)	Yes	Yes
Docking	No	Yes

Fig. 5-38 Comparison of Baseline and Low-Cost SEO Subsystem Modes

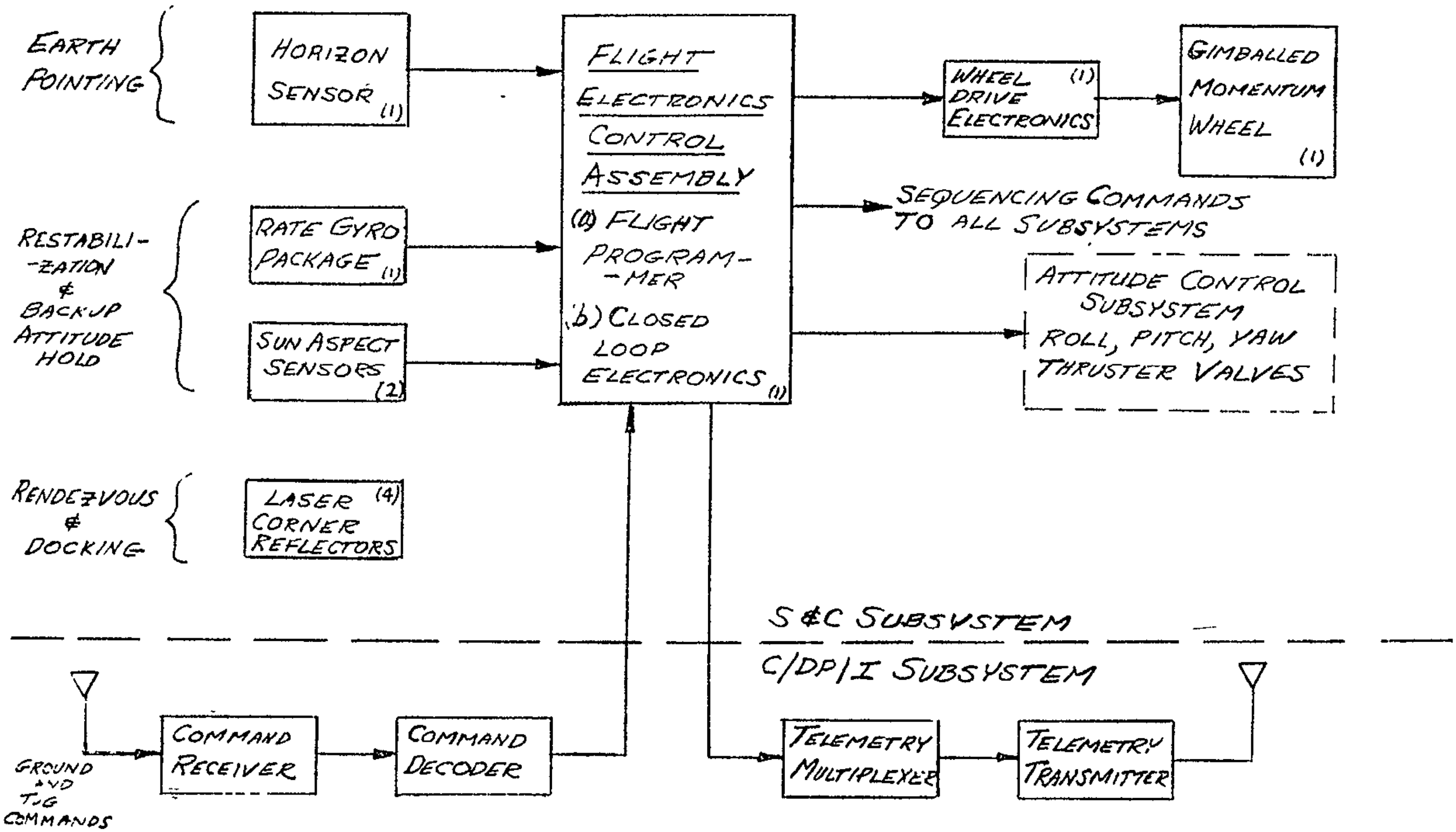


Fig. 5-39 Low-Cost SEO Stabilization and Control Subsystem

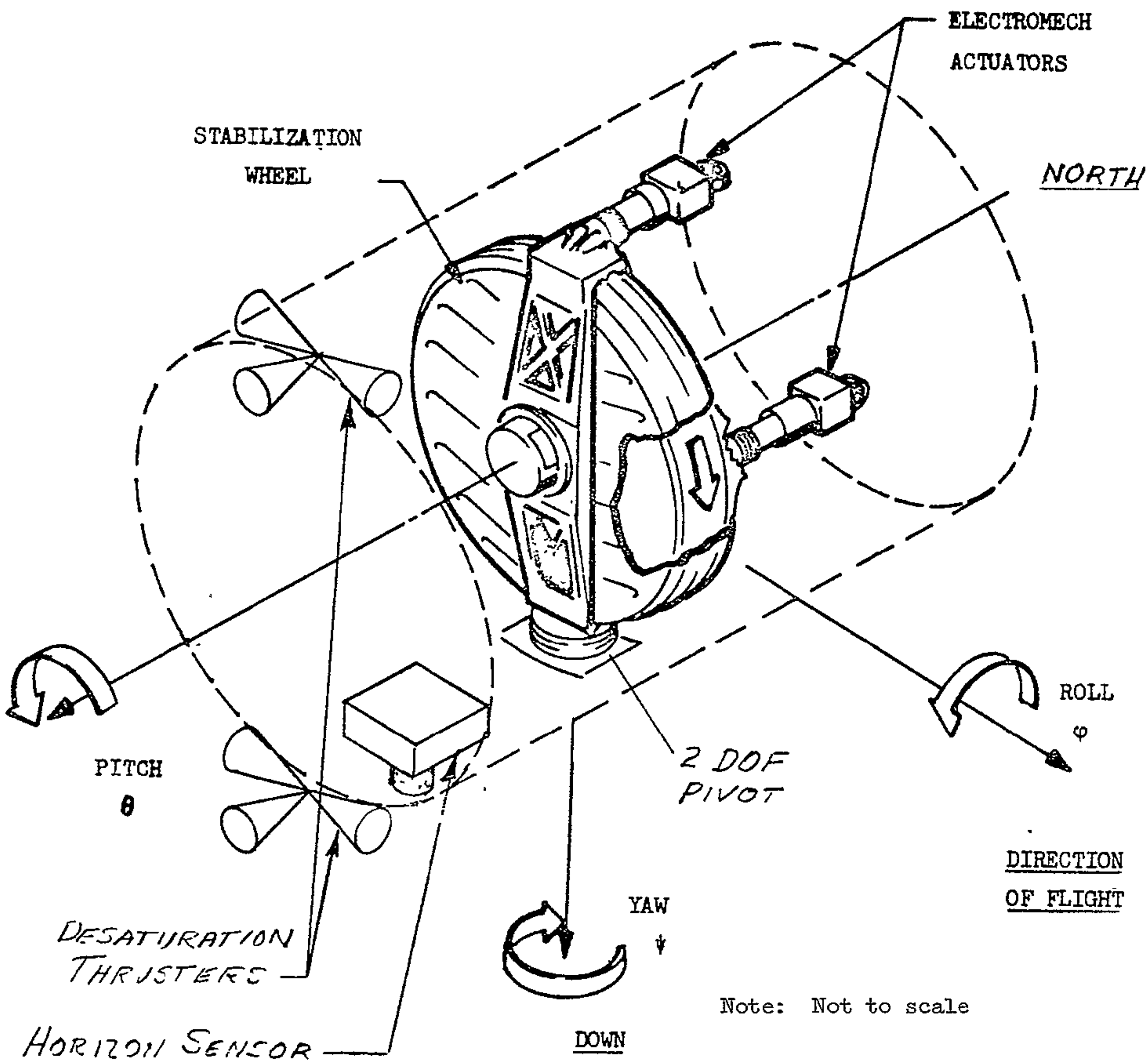
The double-gimballed reaction wheel control system (Fig. 5-40) is a new concept now being applied to three-axis, long-life satellite attitude control. The system uses a single biased angular momentum reaction wheel, mounted on a restricted angle, two degree-of-freedom pivot arrangement for angular momentum control in roll, pitch, and yaw. The low-cost stabilization system requires no yaw sensor.

Cost reductions were accomplished in the design of the low-cost SEO S&C subsystem by the following means:

- Combining the redundant Flight Control Electronics units of the baseline SEO S&C subsystem into a single package
- Using 1970 state-of-the-art off-shelf components in lieu of more expensive Lunar Orbiter components
- Substituting a gimballed momentum wheel for three reaction wheels
- Eliminating 2 Polaris Star Trackers (gimballed momentum wheel eliminates requirement for yaw sensing)
- Eliminating the Flight Electronic Switching Unit (used on Lunar Orbiter for pyrotechnic initiation)
- Repackaging all electronic assemblies using low-cost techniques.
- Modularization of subsystem equipment

Additional details of the S&C subsystem and supporting analyses are contained in LMSC Engineering Memo, PE-22.

5.3.4.5 Communications, Data Processing and Instrumentation (CDPI) Subsystem. The Communication, Data Processing and Instrumentation (CDPI) subsystems of the baseline SEO and the low-cost SEO are functionally identical and the following descriptions are applicable to both.



Note: Not to scale

Fig. 5-40 Dual-Gimballed Wheel System

The functions of the SEO CDPI subsystem are as follows:

(a) transmit photographic, vidicon, spacecraft performance, and command verification data from the spacecraft to ground stations.

(b) receive and decode commands transmitted from ground stations to the SEO and to temporarily store the received commands for verification purposes.

(c) condition all spacecraft data other than video data prior to multiplexing/encoding and to multiplex/encode all non-video data prior to telemetering to ground stations.

(d) sense certain conditions and parameters indicative of the Space subsystems' performance.

The SEO CDPI subsystem includes the following major equipment: An S-band transponder, a tape recorder, a high-gain antenna, a traveling wave tube amplifier, a modulation selector, a command decoder, a low-gain antenna, a PCM Multiplex-Encoder, and pressure and temperature sensors.

The block diagram of the subsystem is presented in Fig. 5-41.

Digital and analog instrumentation signals representing spacecraft equipment status are sent to the Multiplexer/Encoder to be combined with signals from the environmental sensors and command verification data.

The Multiplexer/Encoder consists of an analog multiplexer and A-D converter to digitize 78 analog channels to 8-bit accuracy and interleave the resulting digital data with the data of the digital input channels. The output data frame consists of 128 9-bit words including a 43-bit Legendre code for frame synchronization. The data frame is read out serially at a 50 bps rate. The data has a PCM/NRZ-M format and is presented to the Modulation Selector where it is PSK-modulated onto a subcarrier frequency at 30 KHz. The 30 KHz status telemetry subcarrier is combined with either one of the experiment subcarriers to form a baseband signal to phase modulate the carrier frequency.

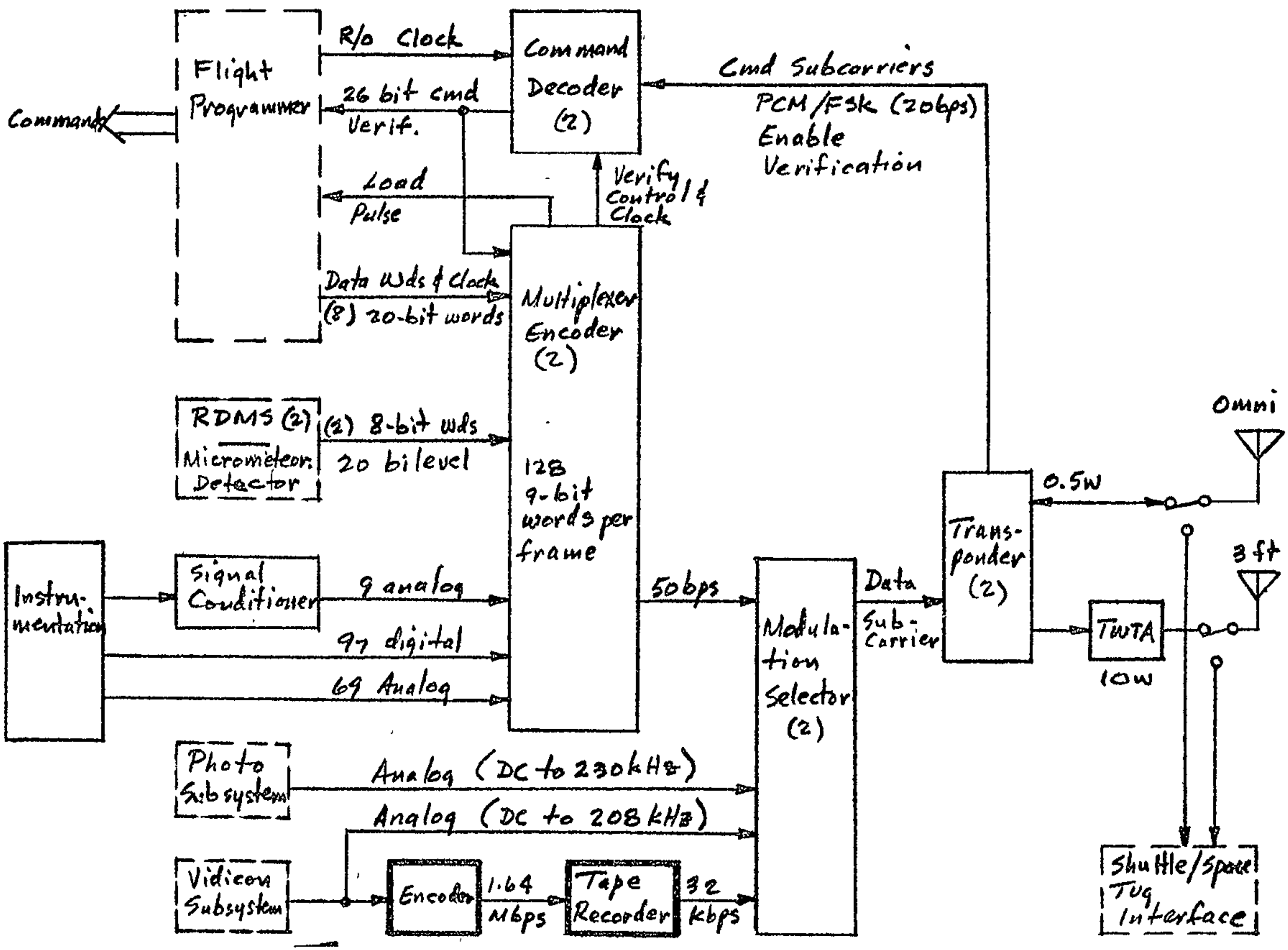


Fig. 5-41 SEO CDPI Subsystem Block Diagram

The Photo Subsystem produces a DC to 230 KHz analog signal. This analog signal is presented to the Modulation Selector where it is vestigial sideband-modulated onto a 310 KHz subcarrier. The video subcarrier and a subcarrier reference, obtained by dividing the 310 KHz subcarrier by 8, is linearly summed with the status telemetry subcarrier during photo readout. The baseband then phase modulates the carrier frequency for transmission.

The Vidicon Subsystem video signal is encoded to 9 bits per picture element to provide a 1.6 Mbps serial bitstream. This data is stored in a digital tape recorder which is required to store up to 36 frames (3.9×10^8 bits total). Readout rate is slowed to 32 Kbps serial PCM. This data is frequency modulated (FM) onto a subcarrier frequency at 150 KHz for transmission sequentially with the photo data. As a backup mode, the analog signal may be modulated onto the photo data subcarrier for realtime readout.

The Modulation Selector selects and modulates data subcarriers under control of the Flight Programmer. Four operational modes are provided. Mode 1 provides telemetry subcarrier only for combination with range code data to modulate the carrier. Mode 2 provides telemetry plus photo data subcarriers to modulate the carrier. Mode 3 provides telemetry only to modulate the carrier for high power transmission. Mode 4 provides vidicon data plus telemetry subcarriers to modulate the carrier.

The transponder receives a carrier frequency in the range of 2110 to 2120 MHz which can be modulated with command subcarriers and range code signals. In the coherent mode, the received carrier is translated up in frequency by the ratio of 240/221 to provide a coherent transmitter frequency. The PRN range code and command subcarriers are demodulated in the receiver portion of the transponder. The range code is combined with the telemetry subcarrier to form a baseband signal which is modulated onto the downlink carrier frequency to provide two-way range tracking. The command subcarriers are presented to the command decoder for command extraction. Phase modulation of the carrier is used to control the spectral distribution of the modulated signal. The modulation index is selected to optimize the power distribution between the carrier and data

subcarriers. Mode 1 uses the 0.5W output of the transponder to drive the omni antenna for transmission of range and telemetry data. The other modes use a portion of the transponder power output to drive the 10W TWT amplifier for transmission via the high gain antenna. The high gain antenna has a 10-degree beamwidth and depends upon the attitude stability of the vehicle to keep it pointed at the ground station.

The command decoder demodulates command subcarriers received from the transponder and stores it temporarily. After verification for validity, the command is routed to the Flight Programmer and is readout to the ground via telemetry.

The principal cost reduction features introduced in the low-cost CDPI subsystem design were (1) the use of low-cost electronic assembly packaging techniques, and (2) the modularization of subsystem equipment into modules which can be separately bench-assembled and completely tested prior to installation into the spacecraft.

A complete description of the low-cost CDPI subsystem and its functional characteristics is provided in LMSC Engineering Memo, PE-23.

5.3.4.6 Electrical Power Subsystem. The Electrical Power subsystems of the baseline SEO and the low-cost SEO are functionally identical and the following descriptions are applicable to them both.

The major equipments for the SEO Electrical Power Subsystem (EPS) are shown in Fig. 5-42. A silicon solar cell array and four secondary nickel cadmium batteries supply continuous-power to the system loads during orbital light and dark periods, respectively. The power system provides unregulated dc power to the bus supplying the spacecraft subsystems. The subsystems condition this dc power as required with individual power supplies. The nominal unregulated bus voltage is 28 volts and may vary from 23 volts to 35 volts, depending on the SEO operating mode.

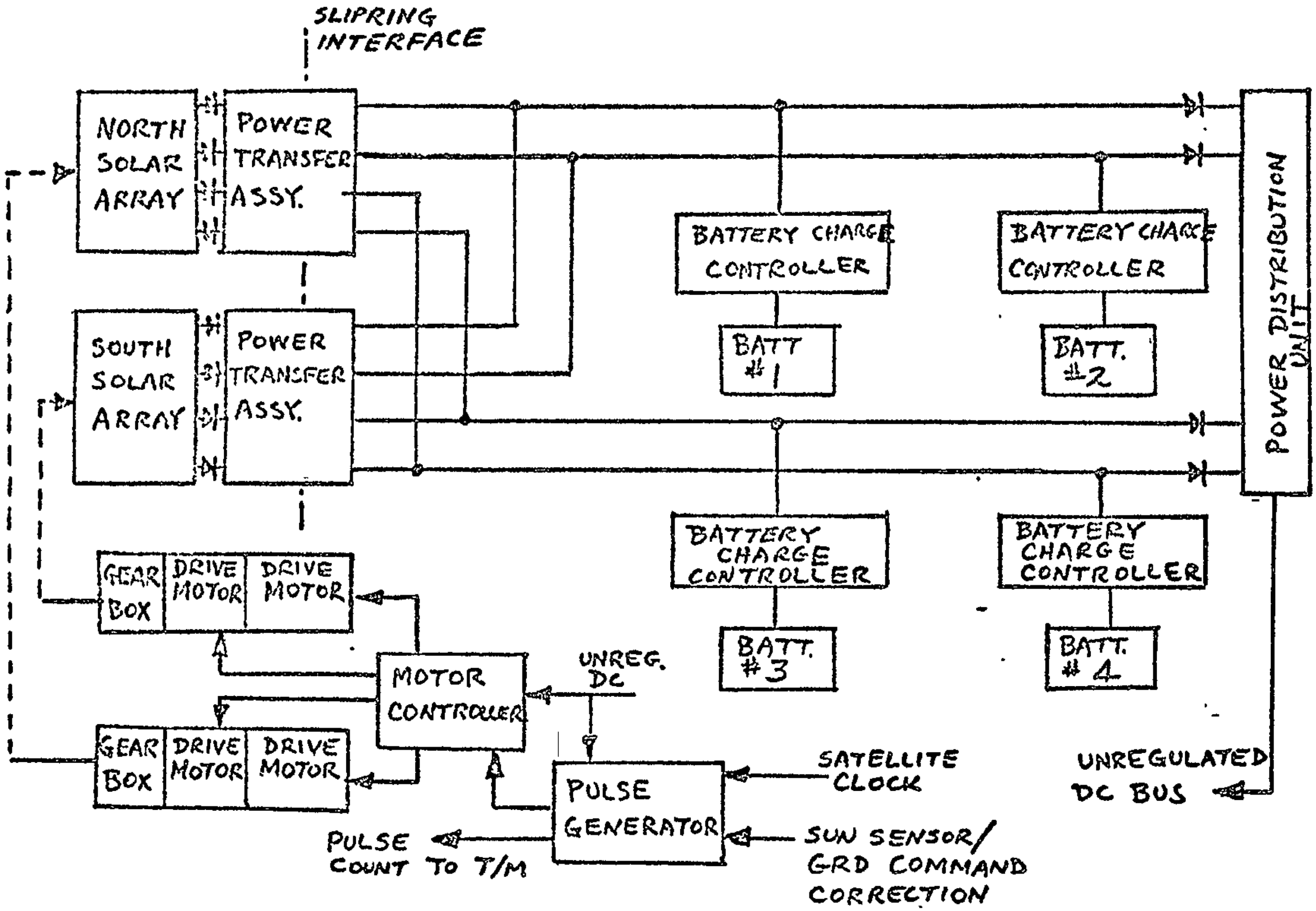


Fig. 5-42 SEO Electrical Power Subsystem

The Electrical Power Subsystem comprises the following major equipments:

a. Solar Cell Array

The solar cell array converts solar energy to electrical energy. The array consists of two paddles with cells on one side. The cells are phosphorous diffused N/P silicon (2 x 2 cm), 12 mils (0.3 mm) thick with 20 mil (0.51 mm) coverglasses. The array contains 10,752 cells. The array operating voltage is 0-50 VDC approximately. The two array paddles are individually driven to track the sun in one axis and a slipring power transfer assembly is provided for each paddle.

b. Voltage Regulator

On-array Zener diode shunt voltage limiters operate to provide an unregulated 24 to 35 vdc to the Power Distribution Unit and the Battery Charge Controllers.

c. NiCd Batteries

NiCd batteries supply power to the spacecraft subsystems during dark periods and aid in supplying peak loads during light periods. The batteries supply power at 23 to 29 volts.

d. State-of-Charge Unit

The state-of-charge unit monitors battery charge and discharge operation using ampere-hour integrators and provides data on battery charge status to the T/M system. The voltage is 23-24 VDC unregulated.

e. Battery Current Shunt Assembly

The battery current shunt assembly measures battery charge and discharge currents and provides proportional outputs to the State of Charge Unit for use in Ampere-Hour integrators.

f. Battery Charge Controllers

Battery charge controllers control battery charging to levels appropriate for the temperature and state of charge of batteries.

g. Tracker Pulse Generator

The tracker pulse generator provides input pulses to the solar array drive motor controller at the fixed rate required for sun tracking. The timing reference used is the SEO clock timer output.

h. Motor Controller

The motor controller provides switching of unregulated 28 vdc to the phases of the stepper motors in proper sequence and for timed duration to rotate the solar array. The time of energizing phases is established by the pulses received from the Tracker Pulse Generator.

i. Tracker Drive Motors

The tracker drive motors step 90 deg when activated by the Motor Controller and provide the force to rotate the solar array, in steps, through a gear box.

j. Power Transfer Assembly

A slipring assembly attached to each rotating solar paddle shaft allows array power and instrumentation signals to be picked up by a brush assembly attached to the vehicle structure. A power transfer assembly is used on each of the two array paddles and continuous 360 deg tracking of the arrays is provided.

k. Power Distribution Unit

The power distribution unit provides the central distribution point for all spacecraft unregulated 28 vdc power, and houses line fuses, current sensors, relays, etc.

In synchronous equatorial orbit, the SEO experiences eclipse shading for from zero to 72 min. for two periods of 44 days each during one year. The EPS operates as follows:

(1) The satellite enters sunlight, with the batteries having supplied all energy requirements during the preceding dark period.

(2) Battery charging commences at the maximum rate and continues until the next dark period.

(3) The satellite enters the dark period, battery charging ceases, and the batteries supply the spacecraft load.

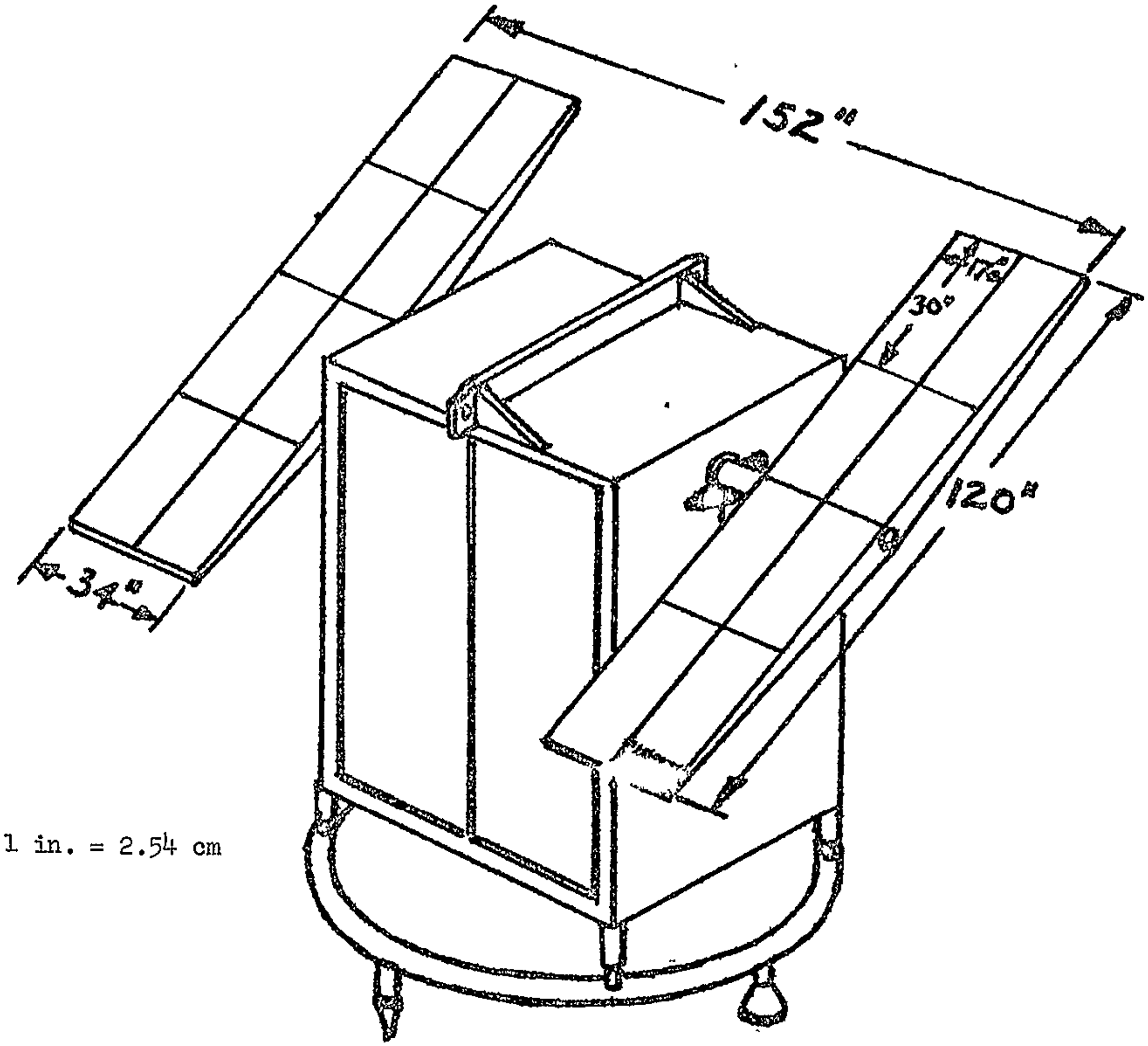
(4) When the eclipse season is over, the batteries are continuously trickle charged.

To reduce its cost, the solar array of the low-cost SEO shown in Fig. 5-43 incorporates the same design concepts employed in the design of the solar array for the low-cost OAO described in the previous subsection. Aluminum sheet metal is used for the paddle structure and the solar cell substrate; and 97.5 percent of the functional solar cells from each manufacturing lot of cells are used. Simplified, low-density packaging is used in the design of charge controllers and other EPS equipment to reduce their cost.

The solar arrays have been made oversize by approximately 10 percent to provide for 4-year degradation rather than 2 years. This allows replacement of solar arrays (for SEO refurbishment) at 4-year intervals and reduces array refurbishment cost by 50 percent.

The subsystem equipment has been modularized to allow easy bench assembly and testing of each module and a minimum of installation time into the spacecraft.

A more detailed description of the low-cost Electrical Power subsystem and its characteristics is provided in LMSC Engineering Memo, PE-24.

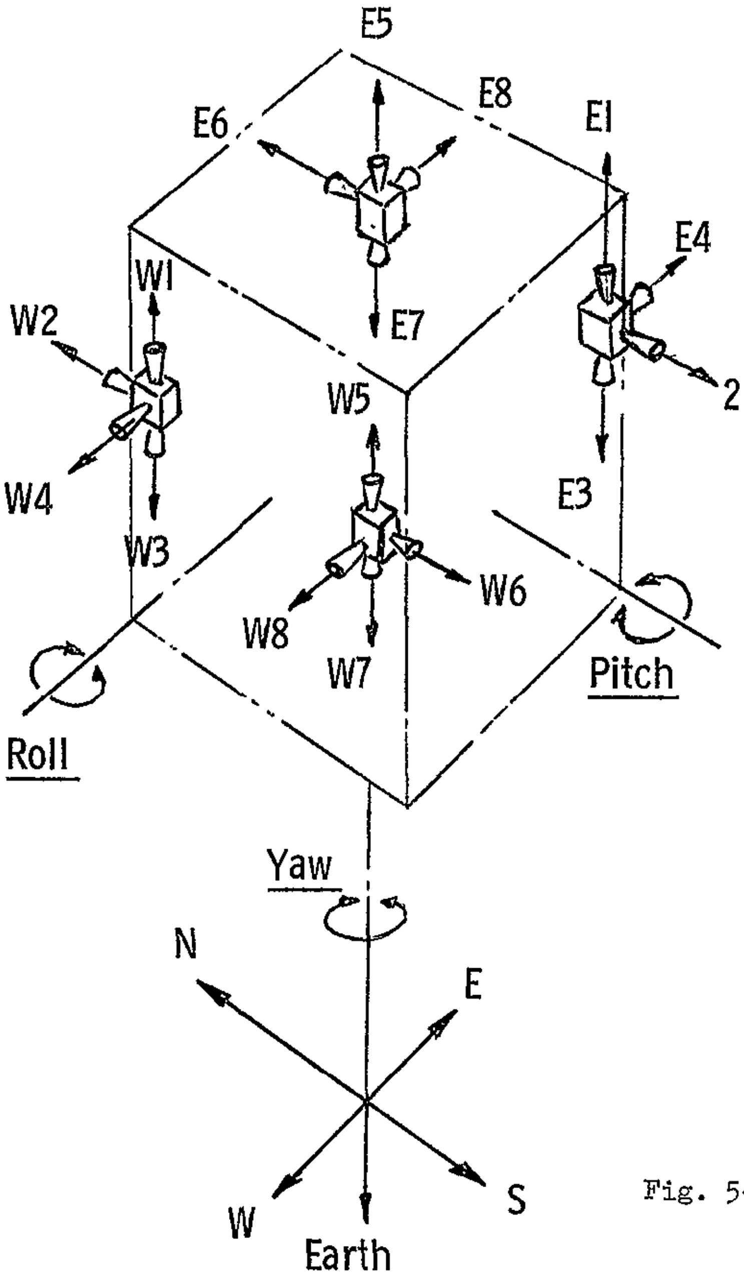


1 in. = 2.54 cm

Fig. 5-43 Low-Cost SEO Solar Array

5.3.4.7 Attitude Control Subsystem. The Attitude Control Subsystem of the low-cost SEO is a cold gas system and provides torques to maintain spacecraft attitude and thrust for East-West translation. It consists of four identical modules, two installed on the "East" side and two on the "West" side of the spacecraft. Each module contains a cluster of four thrusters arranged to obtain 3-axis rotation and East-West translation of the SEO as shown in Fig. 5-44. The schematic diagram of the SEO attitude control module is shown in Fig. 5-45. The propellant storage tank is capable of holding 40 lbs (18.14 kg) of dry Freon 14 at 3000 psi (20.7×10^6 N/m^2) pressure providing a total impulse of 1830 lb-sec (8140 N-sec). The tank is pressurized through a fill valve containing a needle valve which is open to fill or bleed the tank and closed to maintain pressure in the tank. This valve is positioned flush with the vehicle skin so that the tank may be filled with the module installed in the vehicle although the tank is normally filled before the module is installed. A 5 micron filter is connected into the line leading to the solenoid latching valve. Also connected into the line between the tank and the solenoid latching valve is the relief valve. This valve opens and vents gas when pressure raises above 3400 psi (27.6×10^6 N/m^2). A thrust nullifier is installed at the end of the vent line to prevent any thrust being imparted to the vehicle during venting. The solenoid latching valve is normally open and is closed when there is a leak in the downstream system, or when there is an indication that the regulator is malfunctioning causing an abnormal pressure to be supplied to the thrusters. Downstream of the solenoid latching valve is the high pressure regulator which reduces the 3000 psi (20.7×10^6 N/m^2) tank pressure to 60 psi (4.14×10^3 N/m^2) for the thrusters. The final downstream component is the thrust valve cluster consisting of four solenoid valves each attached to a thrust nozzle. Each nozzle produces 0.2 lbs (8.9 N) thrust.

The subsystem was designed so that even with one module failed, the remaining three could provide adequate control thrusting for a 24-month period; this results in a very high operating reliability, 0.999. Because of this, no redundant components were provided in any module.



CONTROL DIRECTION	THRUSTERS
East	W4, W8
West	E4, E8
+ Yaw	W6, E6
- Yaw	W2, E2
+ Pitch	W1, E7
- Pitch	W3, E5
+ Roll	W3, W5
- Roll	W1, W7

Fig. 5-44 Low-Cost SEO Attitude Control Subsystem Thruster System

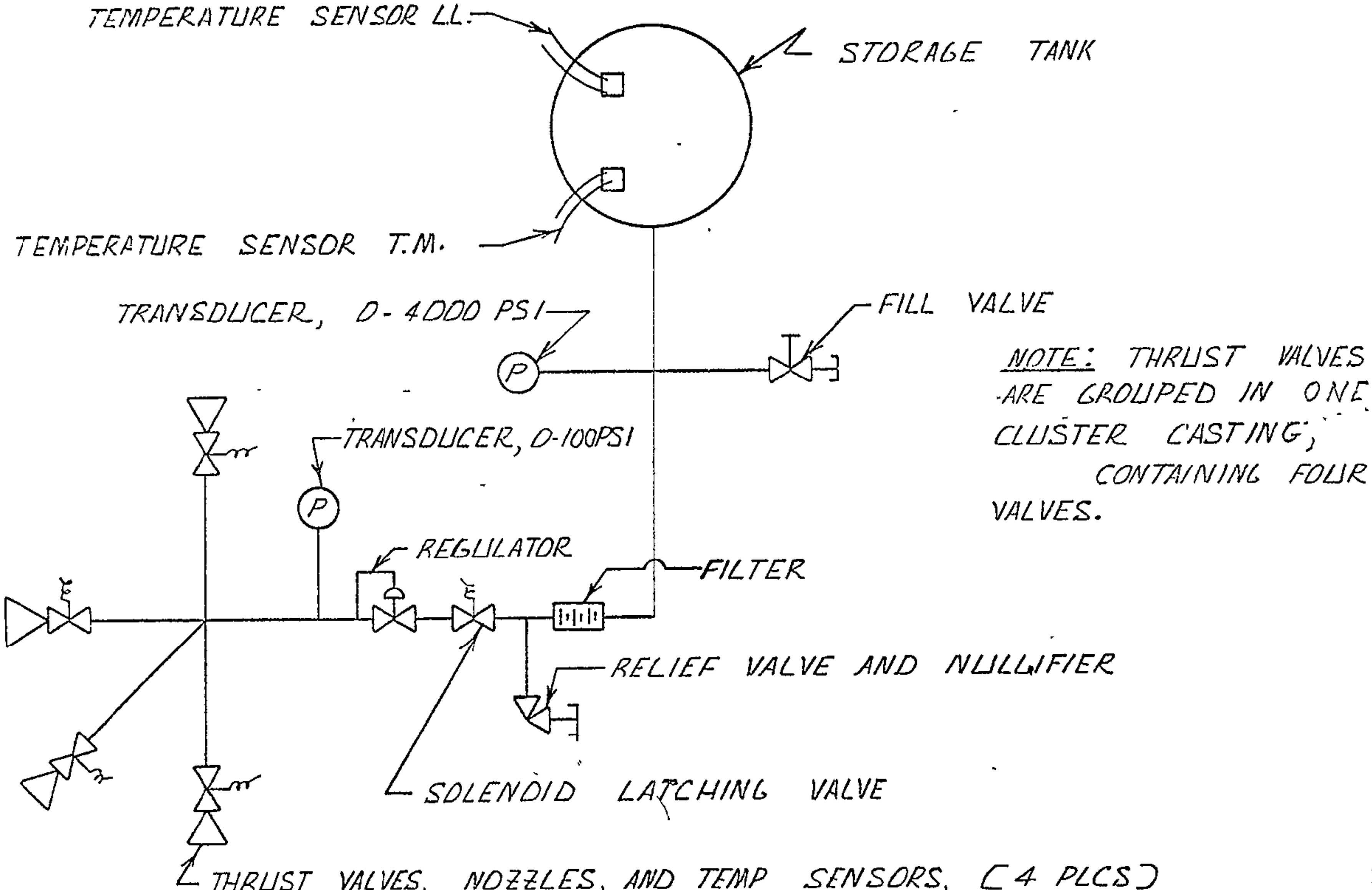


Fig. 5-45 Low-Cost SEO Attitude Control Subsystem Module Schematic

LMSC-A990556

Cost reduction was obtained by:

(1) Using a simple module design which allows complete bench assembly, gas charging, and testing prior to simple insertion into the space rame.

(2) Use of available off-shelf components.

5.3.4.8 Environmental Control Subsystem. The low-cost SEO Environmental Control Subsystem controls the heat loss from the North and South faces of the spacecraft; thermally isolates the top and bottom, and East and West faces; isolates and provides separate thermal control of the Photographic module; and maximizes heat transfer between all other spacecraft systems. The North and South faces of the spacecraft radiate directly to space with some heat inputs from the solar arrays and incident solar radiation at the extreme solar declination angles ($\pm 23.5^{\circ}$). The thermal control surface finish for the North and South faces is selected to radiate the internally dissipated power with and without both solar heating and heating from the arrays. The surface finish of the North and South faces has been selected to dissipate 300 watts of internal power to space and maintain the bulk mean temperature of the spacecraft above $0^{\circ}F$ ($225^{\circ}K$). The surface emittance required is an ϵ of .165 or less. The required solar absorptance has been determined by considering the maximum incident solar radiation on the North and South faces, a maximum view of the solar array at $140^{\circ}F$ ($333^{\circ}K$) and the dissipation of half the power or 150 watts while maintaining the radiating surfaces at about $70^{\circ}F$ ($294^{\circ}K$). The solar absorptance (α_s) must be .394 or less on the North and South face.

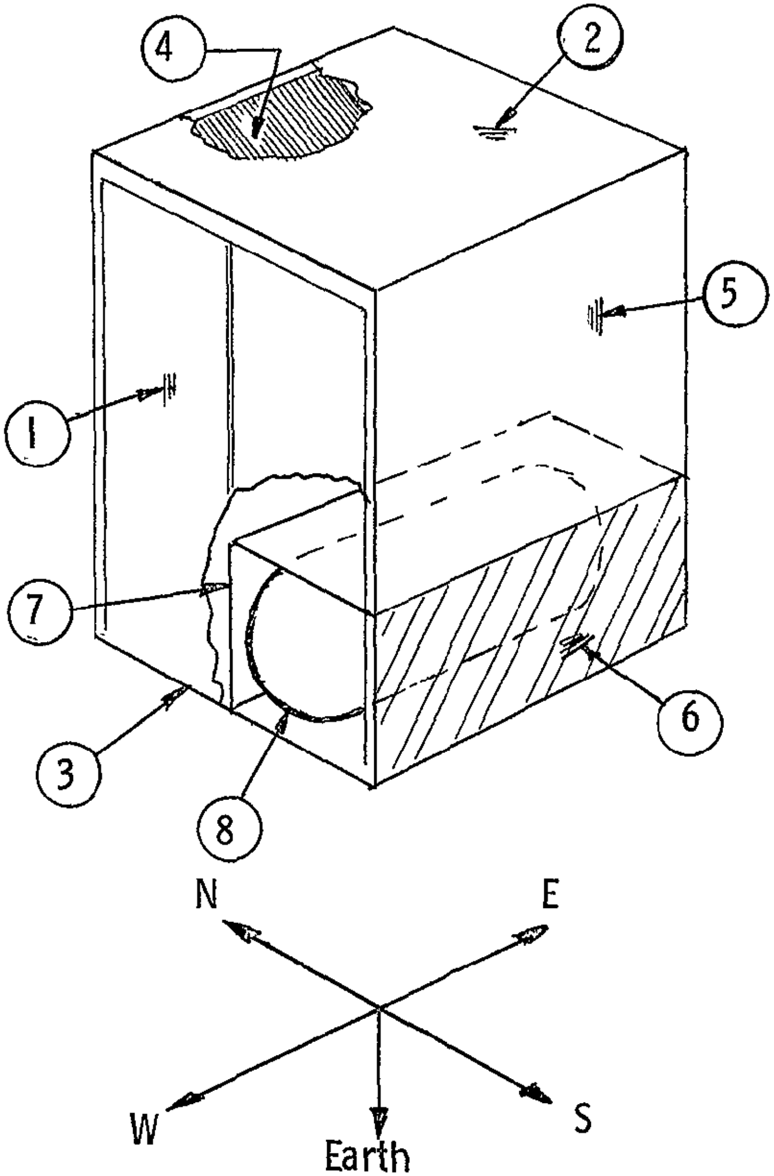
The East and West faces and the top and bottom faces of the spacecraft experience varying solar heat loads and go through a wide range of temperature. Therefore, those faces will be thermally isolated through the use of $\frac{1}{2}$ in. (1.27 cm) of multilayer insulation. The doors on the East and West faces must be designed with consideration for the thermal isolation requirements. The hinges and attachment hardware must be either non-metallic or low-conductivity metals such as titanium or stainless steel.

The internal surfaces will be black to promote radiation heat transfer except for the East and West and top and bottom surfaces which will be either bare aluminum or aluminized. The internal bay structure will be thermally connected to the North and South faces. All electronic modules must be in intimate contact with the mounting surfaces. The equipment module bases must be such that heat transfer to the spacecraft structure is compatible with the temperature control requirements of equipment.

The Photographic module must be thermally isolated from the remainder of the spacecraft by the use of multilayer insulation and low conductance attachment hardware, since the Photographic module desired operating temperature range is 32°F to 40°F (273° to 278°K) and the remainder of the spacecraft temperature range is 0°F to 100°F (255° to 311°K). The external surface finish of that portion of the North face adjacent to the Photographic module must have an α/ϵ of .16/.04 and Optical Solar Reflector (OSR) with an α/ϵ of .06/86 is used to obtain the desired surface properties. The surface is 2/3 Aluminum tape and 1/3 OSR which results in an α/ϵ of .125/.31. The maximum heater power required to maintain the temperature of the Photographic module at 35°F (275°K) with no external heating is 150 watts. A 150 watt heater blanket, thermostatically controlled to 35°F ± 1°F (275 ± 1°K), should be installed inside the Photographic module. The external surface of the cylindrical portion of the Photographic module case should be painted black, and the end domes should be highly reflective clad aluminum. The thermal control design proposed for the Photographic module is feasible with careful design and analysis supported by thermal vacuum testing. Heater controls to ± 1°F (± 1°K) accuracy are within the current state-of-the-art.

The features of the low-cost SEO Environmental Control subsystem are summarized in Fig. 5-46.

5.3.4.9 Summary Weights for Low-Cost SEO. The weight summary, by subsystem, is shown in Figs. 5-47a through 5-47d. The total inert weight, 2,963 lb (1344 kg), compares with the baseline SEO weight of 1,091 lb (494 kg) as shown in the summary on Fig. 5-48.



Spacecraft Temp. - 0^{0}F to 100^{0}F (255^{0} to 311^{0}K)
Photo Module Temp. - 32^{0}F to 40^{0}F (273^{0} to 278^{0}K)

(1) East-West Access Doors - Door interior covered with multilayer insulation; door isolated from spaceframe with stainless steel hinges.

(2)(3) Top and Bottom Sheets - Interior covered with multilayer insulation; thermally isolated by stainless steel attach angles.

(4) Interior Surfaces, Shelves, Partitions - Coated with flat black paint (except those covered with multilayer insulation)

(5) External Surfaces - Coated with aluminum silicone paint (except area on south side covering photo module compartment.)

(6) Photo Module External Sheet - Covered with alternate strips; 2/3 alum. tape, 1/3 optical solar reflector tape (α/ϵ = .125/.31)

(7) Photo Module Compartment - Multilayer insulation on interior surfaces (except outboard south-side panel).

(8) Photo Module - Low-conductance mount to spaceframe; cylindrical surface coated black, ends polished aluminum. 150 watt heater blanket installed internally.

Fig. 5-46 Low-Cost SEO Environmental Control Subsystem

		W/O CONTINGENCY	WITH CONTINGENCY
EXPERIMENT SUBSYSTEM		$\boxed{451\ \text{lb}}$	$\boxed{518\ \text{lb}}$
Photographic Module		365 lb	
Camera	21 lb		
Process Dryer	35		
Optical-Mech. Scanner	65		
Film Handling Assy	46		
Electronics Assy	55		
Camera Internal Supports	20		
N_2 System (incl. 12 lb N_2)	30		
Internal Cabling	4		
Container & Module Base	50		
Film Supply	15		
Bimat Supply	24		
Vidicon Camera Module		75	
Camera Head (2)	26		
Vidicon (2)	6		
Electronics Assy. (2)	14		
Module Base, Cover, Cabling	29		
Radiation Detectors (2)		6	
Micrometeoroid Detectors (20)		5	

1 lb = 0.4536 kg

Fig. 5-47a Low-Cost SEO Weight Breakdown
(1 of 4)

		W/O CONTINGENCY	WITH CONTINGENCY
COMMUNICATIONS, DATA PROCESSING, INSTRUMENTATION SUBSYSTEM		221 lb	254 lb
Data Handling Module (2)		140	160
Electronic components (2 sets)	74 lb		
Module Base, cover, cabling (2)	66		
Communication Module		75	86
Electronic components	44		
Module base, cover, cabling	31		
High-Gain Antenna		4	5
Low-Gain Antenna		2	3
STRUCTURE & MECHANISM SUBSYSTEM		674 lb	742 lb
Structure Assy. (less doors)		342	376
Module Hold-Down Devices (16)		96	106
Doors (4)		142	156
Docking Ring Assy.		94	104
ENVIRONMENTAL CONTROL SUBSYSTEM		61 lb	73 lb
Multilayer insulation		21	25
Surface Coatings		40	48

Fig. 5-47b Low-Cost SEO Weight Breakdown
(2 of 4)

		W/O CONTINGENCY	WITH CONTINGENCY
STABILIZATION & CONTROL SUBSYSTEM		194 lb	223 lb
Sensing & Flight Control Module		85	98
Flight Control Electronics	40 lb		
Horizon Sensor (2)	12		
Solar Aspect Sensor (2)	1		
Module Base, Cover, Cabling	32		
Momentum Module		109	125
Gimballed Momentum Wheel	40		
Wheel Safety Shield	25		
Wheel Drive Electronics	5		
Rate Gyro Package	8		
Module Base, Cover, Cabling	31		
ATTITUDE CONTROL MODULE (4)		500 lb	573 lb
Tank (4) (41 lb Freon cap'y)	284		
Thrust Valve Cluster/Nozzles (4)	14		
Valves (4 sets)	20		
Plumbing (4)	12		
Internal Structure (4)	48		
Transducers/Sensors (4 sets)	6		
Module Base, cover, cabling (4)	116		

Fig. 5-47c Low-Cost SEO Weight Breakdown (3 of 4)

		W/O CONTINGENCY	WITH CONTINGENCY
ELECTRICAL POWER SUBSYSTEM		580 lb	664 lb
Power Control Module		92	105
Electrical Components	56 lb		
Module Base, Cover, Cabling	36		
Battery Module		154 lb	168
Batteries (4)	120		
Shunt	2		
Module Base, Cover, Cabling	32		
Solar Array Paddle (2)		120	138
Solar Array Panels (16) & Interconnects	90		
Solar Paddle Structure (2)	30		
Paddle Drive Module (2)		68	78
Drive Motor (4)	36		
Gear Box (2)	6		
Module Base, Cover, Cabling (2)	26		
Power Transfer Assembly (2)		12	14
90° Gear Box (2)		4	5
Interconnect Elec. Harnesses		130	156
TOTAL PAYLOAD INERT		2681 lb	2963 lb
EXPENDABLES (FREON 14)		164	164
TOTAL PAYLOAD FLIGHT WEIGHT		2845 lb (1311 kg)	3127 lb (1447 kg)

Fig. 5-47d Low-Cost SEO Weight Breakdown (4 of 4)

HARDWARE ELEMENT	BASELINE SEO	LOW-COST SEO**
EXPERIMENT PACKAGE*	294 lb	518 lb
STRUCTURE & MECHANISMS	133	742
ELECTRICAL POWER	312	580
ATTITUDE CONTROL	70	573
STABILIZATION & CONTROL	136	223
COMMUNICATIONS, DATA PROCESSING, & INSTRUMENTATION	147	254
ENVIRONMENTAL CONTROL	11	73
TOTAL DRY WEIGHT	1091 LB	2963 LB
ATTITUDE CONTROL GAS (FREON 14)	60	164
TOTAL PAYLOAD WEIGHT	1151 LB (522 kg)	3127 LB (1447 kg)

* Including 12 lb N_2

** Including weight contingency of approx. 15%

1 lb = 0.4536 kg

Fig. 5-48 Weight Summary - Low-Cost OAO (Shuttle Launched)

5.3.4.10 Reliability of Low-Cost SEO. The ground rule of the study was to design a low-cost SEO with performance equivalent to the Baseline SEO. As shown on Fig. 5-49a, the reliability of the Lunar Orbiter, from which the SEO was extrapolated, was 0.729. By analysis and conversion, but using the same component reliabilities as were derived by Boeing for the Lunar Orbiter, a 1-year baseline figure of 0.778 was derived. These numbers were later converted directly to equivalent 2-year mission-duration figures and a product of 0.547 was established initially for a 2-year baseline SEO.

Actual analysis of duty cycles and failure rates in four subsystem areas (photographic module, stabilization & control, electrical, and communications) resulted in modified reliability estimates, particularly in the photographic module. A "modified 2-year baseline" product of 0.606 resulted.

Finally, the actual subsystem reliabilities for the low-cost SEO were calculated; the results are shown in the last column of Fig. 5-49a. A product reliability figure of 0.600 was established. The principal change was the increased reliability of the Attitude Control subsystem resulting from thruster combinations and modularization.

The basic reasons for reliability change between initial and modified baseline and between modified baseline and low-cost are listed on Fig. 5-49b.

5.3.5 Expendable-Launched SEO Performance and Design Requirements

The low-cost expendable-launched SEO has been designed in accordance with the requirements of LMSC-A981600-A, "General Specification - Performance and Design Requirements for Low Cost Synchronous Equatorial Orbiter (Earth Resources Satellite)", dtd 5 May 1971 (Revised). It is to be launched by an expendable launch vehicle, the Titan IIID/Centaur.

The mission of the expendable-launched SEO is the same as that of the Shuttle-launched SEO defined previously.

SUBSYSTEM	LUNAR ORBITER	1-YEAR BASELINE	2-YEAR INITIAL BASELINE	2-YEAR MODIFIED BASELINE	2-YEAR LOW-COST
Experiment	.899	.845	.841 △4 →	.949	.949
Camera	(.908)	(.904)		(.965)	(.965)
Vidicon	-	(.935)		(.999)	(.999)
Secondary Experiments	(.990)	(.990)		(.984)	(.984)
Structures	.996	.996	.997	.997	.999
Environmental Control	.999	.999	.999	.999	.999
Propulsion	.987	.998	-	-	-
Attitude Control		.992	.961	.961 △3 →	.999
Electrical	.990	.968	.902 △2 →	.871	.871
Stabilization & Control	.882	.981	.862 △1 →	.870 △6 →	.852
Communications, Data Processing, Instrumen.	.945	.980	.886 △2 →	.882 △5 →	.855
Payload Total	.729	.778	.547	.606	.600

Note: See Fig. 5-49b for definition of flag detail △→

Fig. 5-49a Comparison of Subsystem Reliability SEO Payload

REASONS FOR RELIABILITY CHANGES:

1. Inertial Reference Unit deleted. (Burner II upper stage used for injection stabilization and control)

2. Refined redundancy modes

3. Addition of cold gas tanks and modularization of subsystem into four modules; thrusters arranged to allow adequate operation for 2 years even with one of four modules not operating.

4. Duty cycle reductions

5. Eliminate component redundancy; use higher-reliability TWTA

6. Eliminated star trackers; substituted gimballed momentum wheel for three reaction wheels.

Fig. 5-49b Comparison of Subsystem Reliability - SEO Payload

The expendable-launched SEO will be mated with the Titan IIID/Centaur and will be protected during ascent by the Centaur Standard Shroud. The Centaur will transport the SEO to synchronous equatorial orbit, place it on station in operational attitude, and then separate from it and depart.

5.3.6 Low-Cost Expendable-Launched SEO Configuration

The configuration of the expendable-launched low-cost SEO is identical to that shown for the Shuttle-launched SEO except that the docking ring will be replaced by a separation ring incorporating LMSC "Super Zip". Figure 5-50 shows the configuration. The four posts shown at the lower four corners of the SEO will be attached to the upper angle shown in Section A-A.

5.3.7 Description of Low-Cost Expendable-Launched SEO

5.3.7.1 Subsystems of the SEO. The designations of the subsystems of the expendable-launched SEO are the same as those of the Shuttle-launched SEO. Also, the descriptions of the Shuttle-launched SEO subsystems are, in general, applicable to the expendable-launched SEO subsystems and will not be repeated. Only the differences between the subsystems of the expendable-launched SEO and those of the Shuttle-launched SEO will be described. The differences are summarized in Fig. 5-51.

5.3.7.2 Structures and Mechanisms Subsystem. The docking ring assembly will be replaced by the separation ring. The spring cartridges attaching the docking ring to the box structure of the SEO will be replaced by fixed posts. The structure and mechanisms of both the expendable-launched and the Shuttle-launched SEO are designed with high factors of safety for expendable launch vehicle loads to minimize design, analysis, manufacturing, and testing costs. The weight penalty incurred is acceptable for both the expendable-launched and Shuttle-launched SEOs.

5.3.7.3 Stabilization and Control Subsystem. The S&C Subsystem is as described for the Shuttle-launched SEO except that the docking reflectors are deleted.

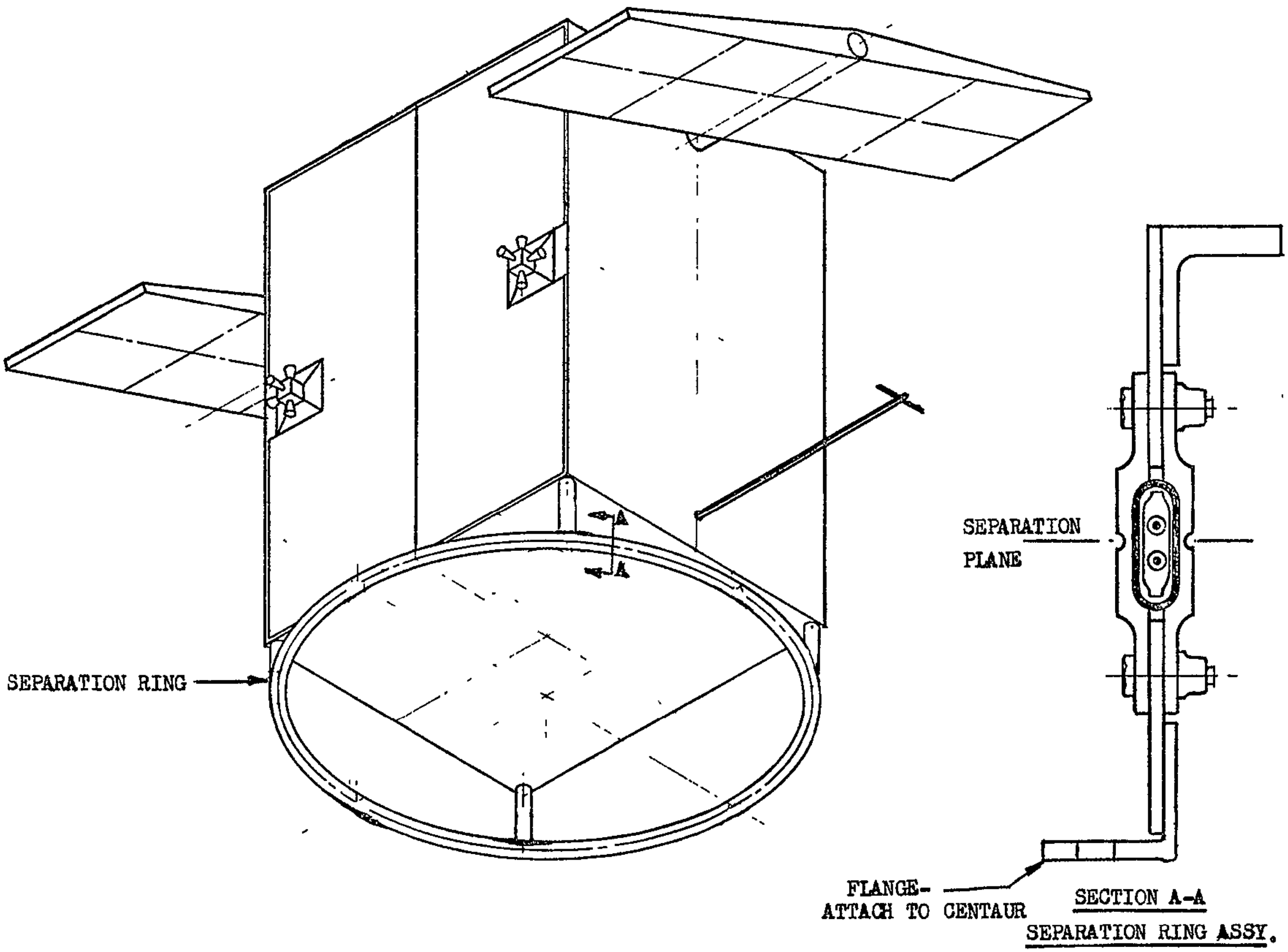


Fig. 5-50 Low-Cost SEO (Expendable-Launched)

ADDED

PAYLOAD INTEGRATION HARDWARE

- Adapter - Payload to Centaur
- Exit Fairing - Centaur "Long"

COMMUNICATIONS, DATA PROCESSING, INSTRUMENTATION

- Transponder (3 in lieu of 2)
- TWTA (2 in lieu of 1)

STRUCTURE

- Separation Ring

DELETED

STABILIZATION & CONTROL

- Docking Reflectors (4)

STRUCTURE

- Docking Ring

Fig. 5-51 Equipment Changes to Low-Cost SEO for Expendable-Booster Launch

5.3.7.4 Communication, Data Processing, and Instrumentation Subsystem. The subsystem described for the Shuttle-launched SEO is augmented by a third transponder and a second traveling wave tube amplifier to increase confidence in the success of the mission; the confidence afforded by in-orbit checkout prior to launch, provided by the Shuttle, is not available.

5.3.7.5 Summary Weights of the Low-Cost Expendable-Launched SEO. The weight summary, by subsystem, for the low-cost expendable-launched SEO is shown on Fig. 5-52.

5.3.7.6 Reliability. Becuase the ground rule of the study was to design a low-cost SEO with equivalent performance to the baseline SEO, very little effort has been expended toward further refinement of the baseline SEO subsystem-level reliabilities. Figure 5-53 summarizes the reliability data. Because of the added redundancy of the TWTA and the Transponder in the CDPI subsystem, the reliability of that subsystem has been improved from .882 to .891. As in the Shuttle-launched SEO, the Attitude Control subsystem reliability has been improved by utilizing four identical modules, with only three of the four required to obtain full controllability of the SEO; the increased reliability is .999 compared to the baseline .961. These two changes result in the payload product reliability improving from the baseline of .606 to .628.

HARDWARE ELEMENT	BASELINE SEO	LOW-COST SEO**
EXPERIMENT PACKAGE*	294 lb	518 lb
STRUCTURE & MECHANISMS	133	722
ELECTRICAL POWER	312	580
ATTITUDE CONTROL	70	573
STABILIZATION & CONTROL	136	223
COMMUNICATIONS, DATA PROCESSING, INSTRUMENTATION	147	277
ENVIRONMENTAL CONTROL	11	73
TOTAL DRY WEIGHT	1091 lb	2966 lb
ATTITUDE CONTROL GAS (FREON 14)	60	164
TOTAL PAYLOAD WEIGHT	1151 lb	3130 lb
PAYLOAD INTEGRATION HARDWARE (ADAPTER)		265
TOTAL LAUNCH WEIGHT (LESS EXIT FAIRING)		3395 lb

* Including 12 lb N_2
** Including weight contingency of approx. 15%

1 lb = 0.4536 kg

Fig. 5-52 Weight Summary - Low-Cost SEO (Expendable-Booster Launched)

SUBSYSTEM	2-YEAR MODIFIED BASELINE SEO	LOW-COST SEO
EXPERIMENT	.949	.949
STRUCTURES	.997	.999
ENVIRONMENTAL CONTROL	.999	.999
ATTITUDE CONTROL	.961	.999
ELECTRICAL	.871	.871
STABILIZATION & CONTROL	.870	.852
COMMUNICATIONS, DATA PROCESSING, INSTRUMENTATION	.882	.891
	$\boxed{.606}$	$\boxed{.628}$

Fig. 5-53 Subsystem Reliability - Expendable-Launched SEO

5.4 LOW-COST SMALL RESEARCH SATELLITE (SRS) DESIGN

The designs of a low-cost Small Research Satellite (SRS), for launch by the Space Shuttle, and by expendable launch vehicles are derived from the Air Forc P-11 satellite (designed and developed by LMSC) and are described in the follo ing paragraphs.

5.4.1 Derivation of the Baseline SRS

The P-11 spacecraft is designed to be mounted on the aft equipment rack of an Agena vehicle. After the Agena attains orbit, the P-11 separates and is injec ted into its mission orbit. The P-11 spacecraft includes structure, a solar power system, a command system, a data system, and a propulsion system. Space weight, power, and data handling capability are provided for various types of payloads. The P-11 and its launcher assembly are shown in Fig. 5-54.

The P-11 as depicted does not provide sufficient electrical power to perform the HiGlo mission. Also the allocation of equipment to subsystems differs slightly from that which has been established as standard for the Payload Ef- fects study. Therefore, a corrected baseline SRS configuration has been cre- ated. The equipment list for the corrected baseline SRS may be found in Figs. 5-55a to 5-55f, together with equipment lists for the Initial Baseline SRS (P-11), the Low-Cost SRS (Expendable launched), and the Low Cost SRS (Shuttle launched). The major difference in configuration between the corrected Base- line SRS and the Initial Baseline SRS is the addition of an identical solar module frame to each of the six frames shown in Fig. 5-54.

5.4.2 Shuttle-Launched SRS Performance and Design Requirements

The design of the low-cost Shuttle-launched SRS has been developed in accord- ance with the requirements of LMSC-A981647-A, "General Specification, Perform- ance and Design Requirements for Low-Cost Small Research Satellite (SRS)", dtd 5 May 1971 (Revised). Specifically the design of the low-cost SRS is to be compatible with the requirements of the HiGlo experiment and to be readily

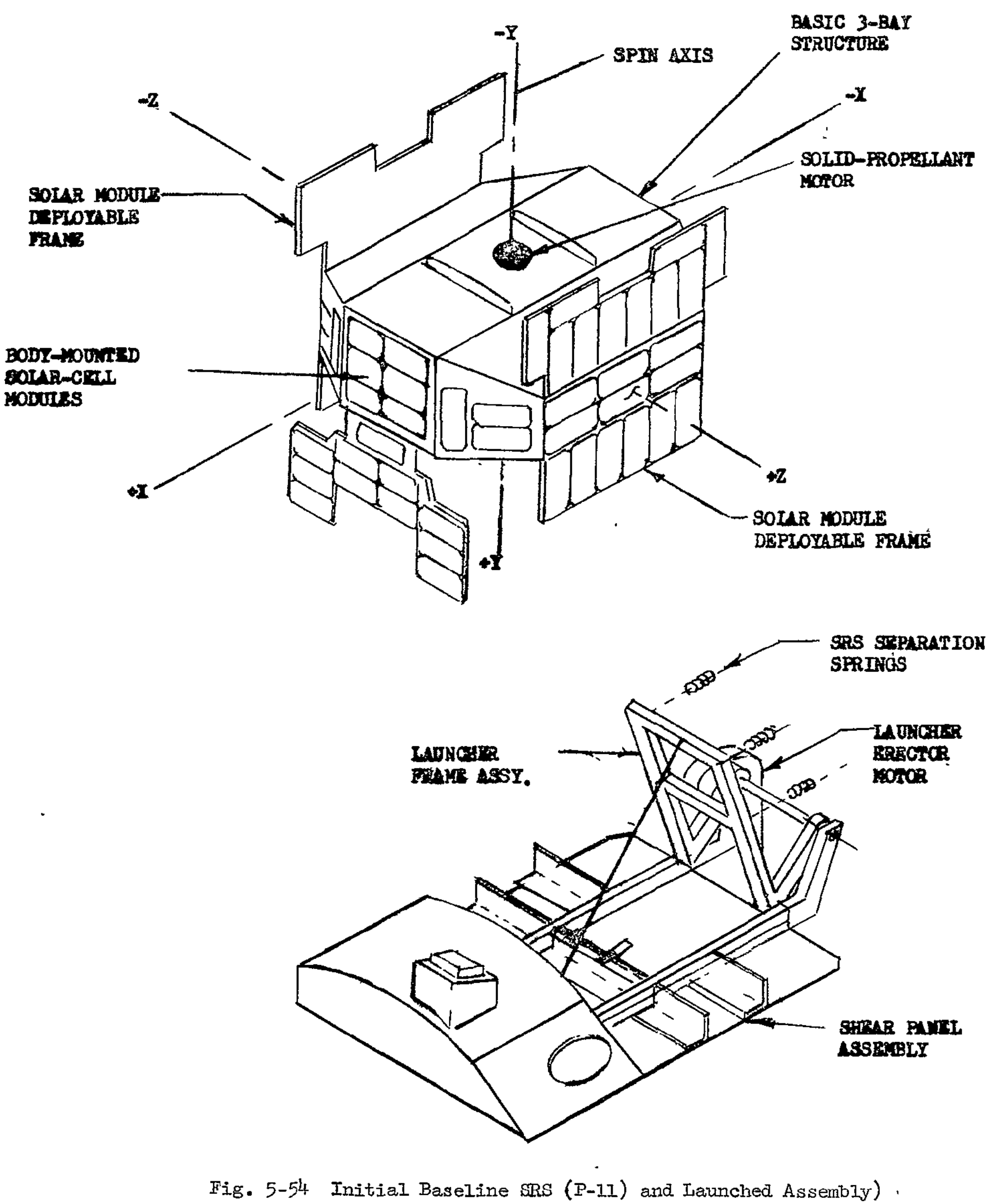


Fig. 5-54 Initial Baseline SRS (P-11) and Launched Assembly)

Hardware Element	Initial Baseline		Corrected Baseline		Low-Cost (Expendable)		Low-Cost (Shuttle)	
	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.
Experiment Installation	(1)	(55.7)	(1)	(55.7)	(1)	(82.3)	(1)	(82.3)
Experiment Module No. 1	-	-	-	-	(1)	(37.4)	(1)	(37.4)
Ion Energy Analyzer (-1)		3.7	1	3.7	1	3.7	1	3.7
Ion Energy Analyzer (-2)	1	1.	1	1.	1	1.	1	1.
Epithermal Electron Anlr. (-1)	1	3.5	1	3.5	1	3.5	1	3.5
Epithermal Electron Anlr. (-2)	2	2.	2	2.	2	2.	2	2.
Cyl. Langmuir Probe	1	2.	1	2.	1	2.	1	2.
0° Multichannel Particle Anlr.	1	2.6	1	2.6	1	2.6	1	2.6
55° Multichannel Particle Anlr.	1	2.2	1	2.2	1	2.2	1	2.2
Proton Hydrogen Anlr.	1	2.8	1	2.8	1	2.8	1	2.8
Electric Field Probe	1	2.6	1	2.6	1	2.6	1	2.6
Cal & Interface Box	1	1.	1	1.	1	1.	1	1.
Data Mode Box	1	1.	1	1.	1	1.	1	1.
Module Base	-	-	-	-	1	10.	1	10.
Module Cables	-	-	-	-	Set	3.	Set	3.

1 lb = 0.4536 kg

Fig. 5-55a SRS Weight Breakdown (No Contingency)

Hardware Element	Initial Baseline		Corrected Baseline		Low-Cost (Expendable)		Low-Cost (Shuttle)	
	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.
Experiment Module No. 2	-	-	-	-	(1)	(44.9)	(1)	(44.9)
Ion Energy Analyzer (-1)	1	3.7	1	3.7	1	3.7	1	3.7
Ion Energy Analyzer (-2)	1	1.	1	1.	1	1.	1	1.
Electrostatic Analyzer	1	5.9	1	5.9	1	5.9	1	5.9
90° Multichannel Particle Analyzer	1	2.8	1	2.8	1	2.8	1	2.8
Total Energy Proton Sensor (-1)	1	1.7	1	1.7	1	1.7	1	1.7
Total Energy Proton Sensor (-2)	1	1.6	1	1.6	1	1.6	1	1.6
Angular Distribution Inst.	3	4.5	3	4.5	3	4.5	3	4.5
ADI Power Supply	1	1.8	1	1.8	1	1.8	1	1.8
Penetrating Rad. Monitor	1	5.9	1	5.9	1	5.9	1	5.9
Tri-Axis Magnetometer	1	1.	1	1.	1	1.	1	1.
TAM Electronics	1	2.	1	2.	1	2.	1	2.
Module Base	-	-	-	-	1	10.	1	10.
Module Cables	-	-	-	-	Set	3.	Set	3.

1 lb = .4536 kg

Fig. 5-55b SRS Weight Breakdown (No Contingency)

LMSC-A990556

Hardware Element	Initial Baseline		Corrected Baseline		Low-Cost (Expendable)		Low-Cost (Shuttle)	
	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.
Electrical Power S/S		(49.0 lb)		(99.8 lb)		(182.9 lb)		(144.9 lb)
Solar Panel Structure Assy	6	4.2	12	12.	-	-	-	-
Solar Cell Modules	90	15.1	204	33.	-	-	-	-
Solar Panel (27.6" x 22")	-	-	-	-	8	51.3	8	51.3
NiCd Battery (6 AH)	1	16.	2	32.	-	-	-	-
NiCd Battery (12 AH)	-	-	-	-	2	60.	1	30.
Battery Dissipator	1	0.6	2	1.2	2	2.	1	1.
Power Control Unit	1	3.5	2	7.	2	14.	1	7.
Pyro Control Unit	1	2.6	1	2.6	1	2.6	1	2.6
Module Base	-	-	-	-	1	10.	1	10.
Module Cables	-	-	-	-	set	3.	set	3.
Electrical Harness	Set	7.	Set	12.	set	40.	Set	40.

1 lb = .4536 kg

Fig. 5-55c SRS Weight Breakdown (No Contingency)

Hardware Element	Initial Baseline		Corrected Baseline		Low-Cost (Expendable)		Low-Cost (Shuttle)	
	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.
CDPI Subsystem		(57.0 lb)		(61.4 lb)		(67.5 lb)		(57.4 lb)
Command Receiver	1	2	2	4	2	2.6	1	1.3
Command Decoder	1	2	1	2	1	3	1	3
Status Commutator (5x60)	1	3	1	3	1	3	1	3
Exp. Commutator (5x90)	-	-	1	3	1	3	1	3
Exp. Commutator (1x90)	-	-	1	3	1	3	1	3
Exp. Commutator (10x30)	-	-	1	2	1	2	1	2
VCO	3	1.5	4	2	4	0.4	4	0.4
PM Transmitter	-	-	1	3.6	1	2	1	2
FM Transmitter	2	7.2	1	3.6	1	2	1	2
Diplexer (Multicoupler)	2	6	1	3	1	3	1	3
VHF Antenna	1	3	2	6	2	1	2	1
Tape Recorder	2	20	2	20	2	20	1	10
Primary Timer	1	5.7	1	5.7	1	7.	1	7
RF Switch	-	-	-	-	-	-	2	1.2
Status Sensors	12	0.5	12	0.5	12	0.5	12	0.5
Earth Horizon Sensor	1	0.8	-	-	-	-	-	-
Solar Aspect Sensor	1	0.9	-	-	-	-	-	-
SAS Electronics	1	2.4	-	-	-	-	-	-
Command Antenna	1	2	-	-	-	-	-	-
Module Base	-	-	-	-	1	10	1	10
Module Cables	-	-	-	-	Set	5	Set	5.

Fig. 5-55d SRS Weight Breakdown (No Contingency)

1 lb = .4536 kg

LMSC-A990556

Hardware Element	Initial Baseline		Corrected Baseline		Low-Cost (Expendable)		Low-Cost (Shuttle)	
	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt
Stabilization & Control Subsystem	1	(2.2 lb)	1	(26.3 lb)		(55.3 lb)		(53.3 lb)
Wobble Damper	1	2.2	1	3	1	10	1	10
Magnetic Torquer	-	-	1	5	1	10	1	10
Magn. Torq. Electronics	-	-	1	2	1	5	1	5
Earth Horizon Sensor	-	-	2	4.	2	4	2	4
Solar Aspect Sensor	-	-	1	0.3	1	0.3	1	0.3
SAS Electronics	-	-	1	3	1	3	1	3
Earth Sensor Electronics	-	-	1	2	1	3	1	3
Aux. Fl. Cont. Elect.	-	-	1	2	1	2	-	-
Module Base	-	-	-	-	1	10	1	10
Module Cables	-	-	-	-	Set	2	Set	2
Propulsion & AC S/S		(46.6 lb)		(27.0 lb)		(48.0 lb)		(48.0)
Spin Motor - Talley 65-004-002	2	2.0	8	8.0	-	-	-	-
Rocket Motor - Thiokol TE 345-11	-	-	-	-	1	(WP=21) 36	1	(WP=21) 36
Rocket Motor - LPC 2P14102	2	(WP=16.4ea) 44.6	1	(WP=13) 19	-	-	-	-
Spin Motor - Hercules SR9-HP-1	-	-	-	-	8	12	8	12

WP = Weight of Propellant
1 lb = .4536 kg

Fig. 5-55e SRS Weight Breakdown (No Contingency)

Hardware Element	Initial Baseline		Corrected Baseline		Low-Cost (Expendable)		Low-Cost (Shuttle)	
	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt.	Qty	Total Wt
Structure & Mechanisms S/S		(40.3 lb)		(40.3 lb)		(180.2 lb)		(180.2 lb)
Structure	1	40.3	1	40.3	1	180.2	1	180.2
Environmental Control S/S		(6.4 lb)		(9. lb)		(40. lb)		(40. lb)
Multilayer Insulation		2.4		5.		20		20.
Surface Coatings		4.		4.		20		20.

1 lb = .4536 kg

Fig. 5-55f SRS Weight Breakdown (No Contingency)

adaptable to the requirements of other space physics missions. For the HiGlo mission the SRS will be carried by the Space Shuttle into a low earth orbit having an apogee of 150 nm (278 km). It will be checked out in the Shuttle cargo bay and then ejected from the Shuttle by spring mechanisms, spun by small rockets and propelled into the orbit [h_p = 150 nm (278 km), h_a = 275 nm (509 km)] by a single solid propellant rocket motor. One HiGlo mission operations plan provides for three complete HiGlo spacecraft to be carried into orbit by the Shuttle and for two of them to be launched. The third spacecraft will be a spare to be launched if either of the other two is found to malfunction during checkout in the Shuttle cargo bay.

For the HiGlo mission, the SRS orbit will be near-polar and sun-synchronous. The spacecraft spin axis will be maintained normal to the orbit plane and the solar cell panels will be perpendicular to the sun's rays.

5.4.3 Low-Cost Shuttle-Launched SRS Configuration

The general configuration of the low-cost Shuttle-launched SRS is shown in Fig. 5-56. The location of equipment modules in the spacecraft is shown in Fig. 5-57.

In designing the low-cost SRS, advantage has been taken of the exceptional payload weight and volume capability of the Space Shuttle to create an SRS that has greater growth potential than conventional spacecraft. Volume and structural support is available for the installation of up to 100 lb (45.4 kg) of additional experiment equipment and a similar quantity of subsystem equipment. The power available for experiment and spacecraft equipment, nominally 50 watts average, can be increased to 100 watts average by the installation of additional solar cell panels in place of plain skin panels. A second propulsion rocket may be installed to provide greater total propulsion impulse for alternate missions.

The low density packaging and ease of access to equipment provided by the low-cost SRS will simplify its conversion to other missions.

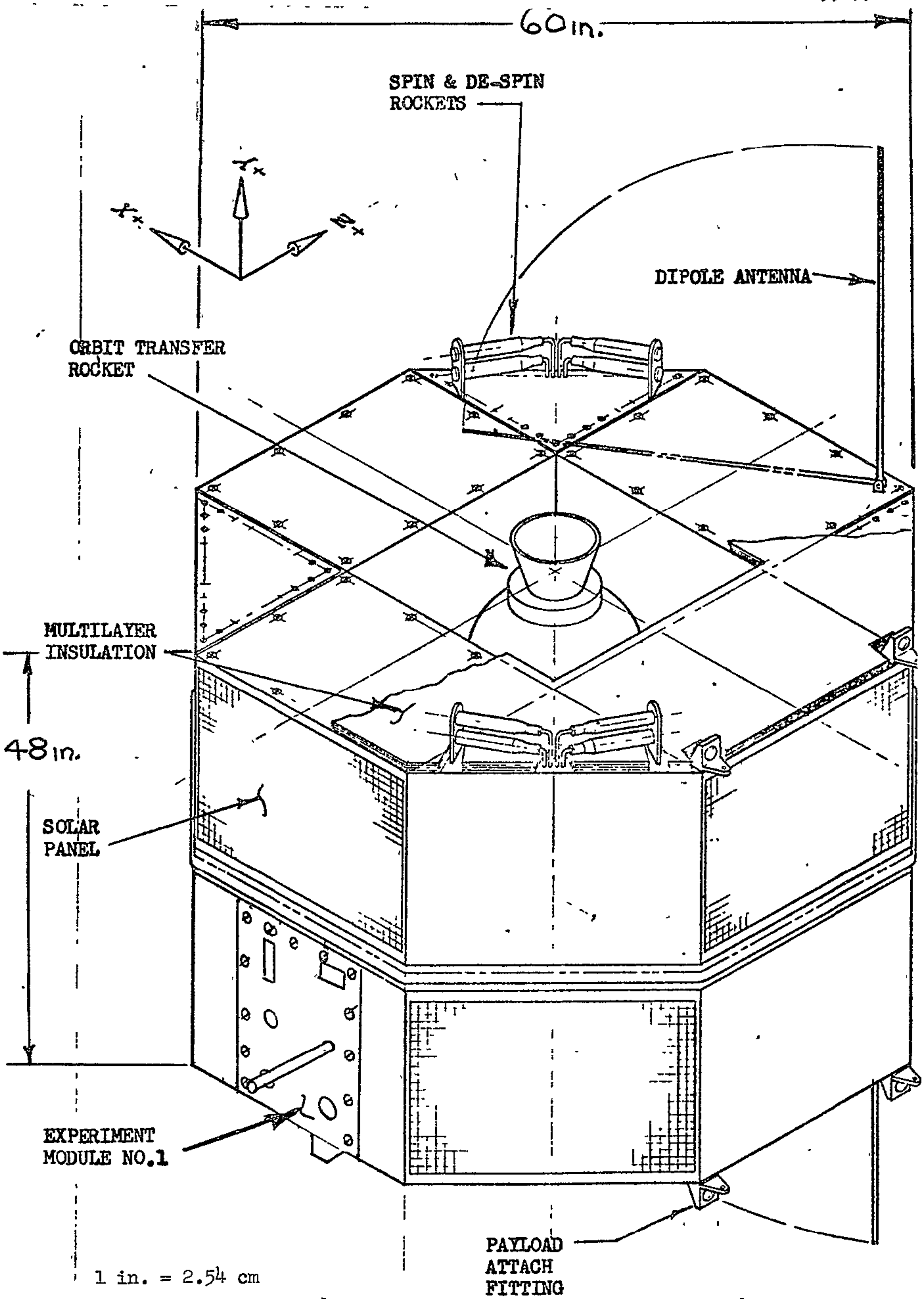


Fig. 5-56 General Configuration - Low-Cost SRS

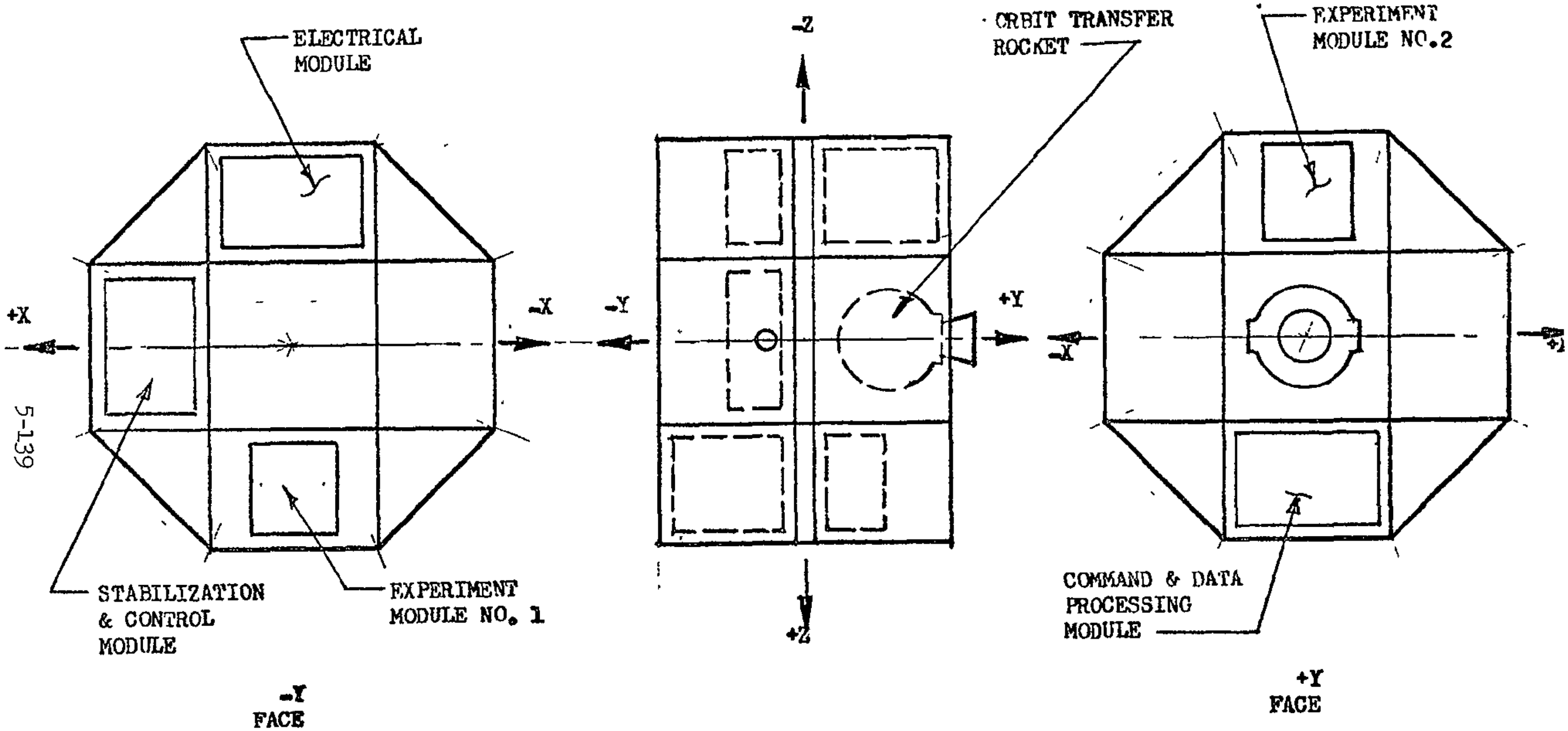


Fig. 5-57 Low-Cost SRS Module Locations

5.4.4 Description of Low-Cost Shuttle-Launched SRS

5.4.4.1 Subsystems of the Low-Cost Shuttle-Launched SRS. The subsystems comprising the low-cost SRS, both Shuttle-launched and expendable-launched, and the LMSC Engineering Memos that describe them in detail are as follows:

Subsystem	LMSC Engineering Memo
Experiment Installation	PE-41
Stabilization & Control	PE-42
Communications, Data Processing & Instrumentation	PE-43
Electrical Power	PE-44
Propulsion & Attitude Control	PE-45
Environmental Control	PE-46
General Payload Description	PE-47

Summary descriptions of all subsystems are presented in the following paragraphs. A separate Engineering Memo describing the Structures & Mechanisms Subsystem has not been prepared and that subsystem is described first.

5.4.4.2 Structures & Mechanisms Subsystem. The primary structure of the low-cost SRS is shown in Fig. 5-58. Four deep beams made up of aluminum sheet and extrusions form an "eggcrate" assembly in which subsystem equipment may be installed. This assembly is strengthened by heavy tee extrusions connecting the outboard mid-points of the deep beams, and by angles connecting the outboard corners of the beams. Shear strength is provided by a heavy shear panel at the mid-plane of the structure, and by bulkheads that close the ends of the four triangular compartments of the "eggcrate". Each of the eight plane surfaces of the octagonal primary structure assembly provides for the attachment of two solar panels or skin panels. As configured for the HiGlo mission, the SRS has eight solar panels, three skin panels incorporating view ports for sensors, and five plain skin panels. The +Y and -Y faces of the SRS structure are closed by stiffened aluminum panels and insulated by multilayer insulation blankets. The structure assembly includes secondary structure for the attachment of equipment modules, rocket motors, antennas, and fittings for handling and deployment of the SRS. The SRS structure has been designed with generous provisions for augmentation of the baseline HiGlo mission. Volume is available for the addition

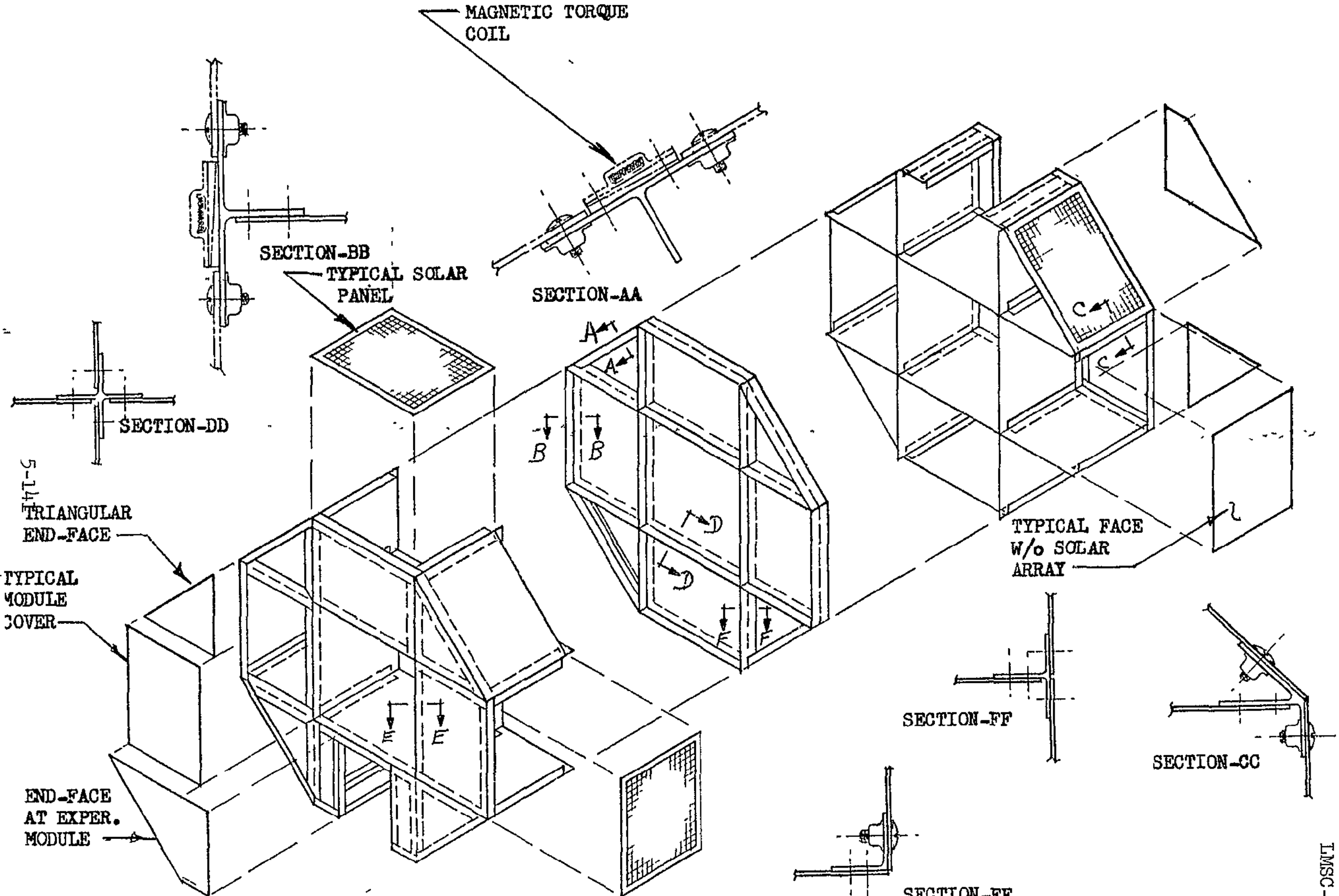


Fig. 5-58 Low-Cost SRS Structural Assembly

of experiment and subsystem equipment including a second propulsion rocket. Also, the average power available for subsystem and experiment equipment can be increased from 50 watts to approximately 100 watts by the installation of additional solar cell panels. The strength of the structure, based on factors of safety of three or greater, permits the addition of equipment without extensive analysis and redesign.

Most of the equipment of the HiGlo SRS is packaged in readily removable modules to reduce development and test costs. The location of modules in the SRS was previously shown in Fig. 5-57 and the equipment making up each module is listed in Figs. 5-59a and 5-59b.

Since the SRS is spin stabilized it must be dynamically balanced. During the detail design of the spacecraft consideration will be given to the placement of equipment to minimize unbalance. Balance weights will then be used to establish dynamic balance.

5.4.4.3 Experiment Installation. The experiment chosen for consideration in the design of the low-cost SRS is the HiGlo experiment that was flown on the USAF OV-1-18. The objective of this experiment was to obtain data concerning ion energies and density gradients in the F layer of the ionosphere. The experiment equipment was as follows:

Item	Wt (lbs)	Pwr. Average
Ion Energy Analyzer (4)	7.7 lbs	5.2
Epithermal Electron Analyzers (3)	5.5	4.8
Cylindrical Langmuir Probe	2.0	3
Electrostatic Analyzer	5.9	2.3
0° Multichannel Particle Analyzer	2.6	1
55° Multichannel Particle Analyzer	2.8	1
90° Multichannel Particle Analyzer	2.2	1
Proton Hydrogen Analyzer	2.9	0.9
Total Energy Proton Sensors (2)	3.3	1.4
Angular Distribution Instruments (3) plus power supply	6.3	3.2
Penetrating Radiation Monitor	5.9	2.6
Electric Field Probe	2.6	1.3
3-Axis Magnetometer (Flux gate)	0.6	1
Calibration & Interface Box	1	0.3
Data Mode Box	1	1.1
	52.3 lbs (23.7 kg)	30.1 watts

Subsystem	Module	Equipment in Module	Module Weight (lb)
Experiment Installation	Experiment No. 1	Ion Energy Analyzer (-1) Ion Energy Analyzer (-2) Epithermal Electron Analyzer (-1) Epithermal Electron Analyzer (-2), 2 Cylindrical Langmuir Probe 0° Multichannel Particle Analyzer 55° Multichannel Particle Analyzer Proton Hydrogen Analyzer Electric Field Probe Calibration & Interface Box Data Mode Box Module Base Module Cables	Basic 37.4 15% Cont. 5.6 Total 43.0
Experiment Installation	Experiment No. 2	Ion Energy Analyzer (-1) Ion Energy Analyzer (-2) Electrostatic Analyzer 90° Multichannel Particle Analyzer Total Energy Proton Sensor (-1) Total Energy Proton Sensor (-2) Angular Distribution Instrument, 3 AD1 Power Supply Penetrating Radiation Monitor Tri-Axis Magnetometer TAM Electronics Module Base Module Cables	Basic 44.9 15% Cont. 6.7 Total 51.6
Electrical	Electrical	NiCad Battery, 12 AH Battery Dissipator Power Control Unit Pyro Control Unit Module Base Module Cables	Basic 53.6 15% Cont. 8.0 Total 61.6

1 lb = 0.45359 kg

Fig. 5-59a Low-Cost SRS Modules (Shuttle-Launched)(1 of 2)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Communication, Data Process., & Instrumen.	Communication & Data Processing	Command Receiver Command Decoder Primary Timer Tape Recorder Status Commutator (5 x 90) Experiment Commutator (5 x 90) Experiment Commutator (1 x 90) Experiment Commutator (10 x 30) Voltage Controlled Oscillators, 4 PM Transmitter FM Transmitter Diplexer RF Switch Module Base Module Cables	Basic 55.9 15% Cont. 8.4 Total 64.3
Stabilization & Control	Stabilization & Control	Earth Horizon Sensor, 2 Earth Horizon Sensor Electronics Solar Aspect Sensor SAS Electronics Magnetic Torquer Electronics Module Base Module Cables	Basic 27.3 15% Cont. 4.1 Total 31.4

Note: Following equipment is not part of any module and is mounted directly on Spacecraft structure:

Solar Panel (8 or 13)
Spacecraft Electrical Harness
VHF Antenna (2)
Wobble Damper
Magnetic Torquer Coil

Spin Motor (4)
Despin Motor (4)
Primary Rocket Motor
Thermal Insulation and Coatings

1 lb = .4536 kg

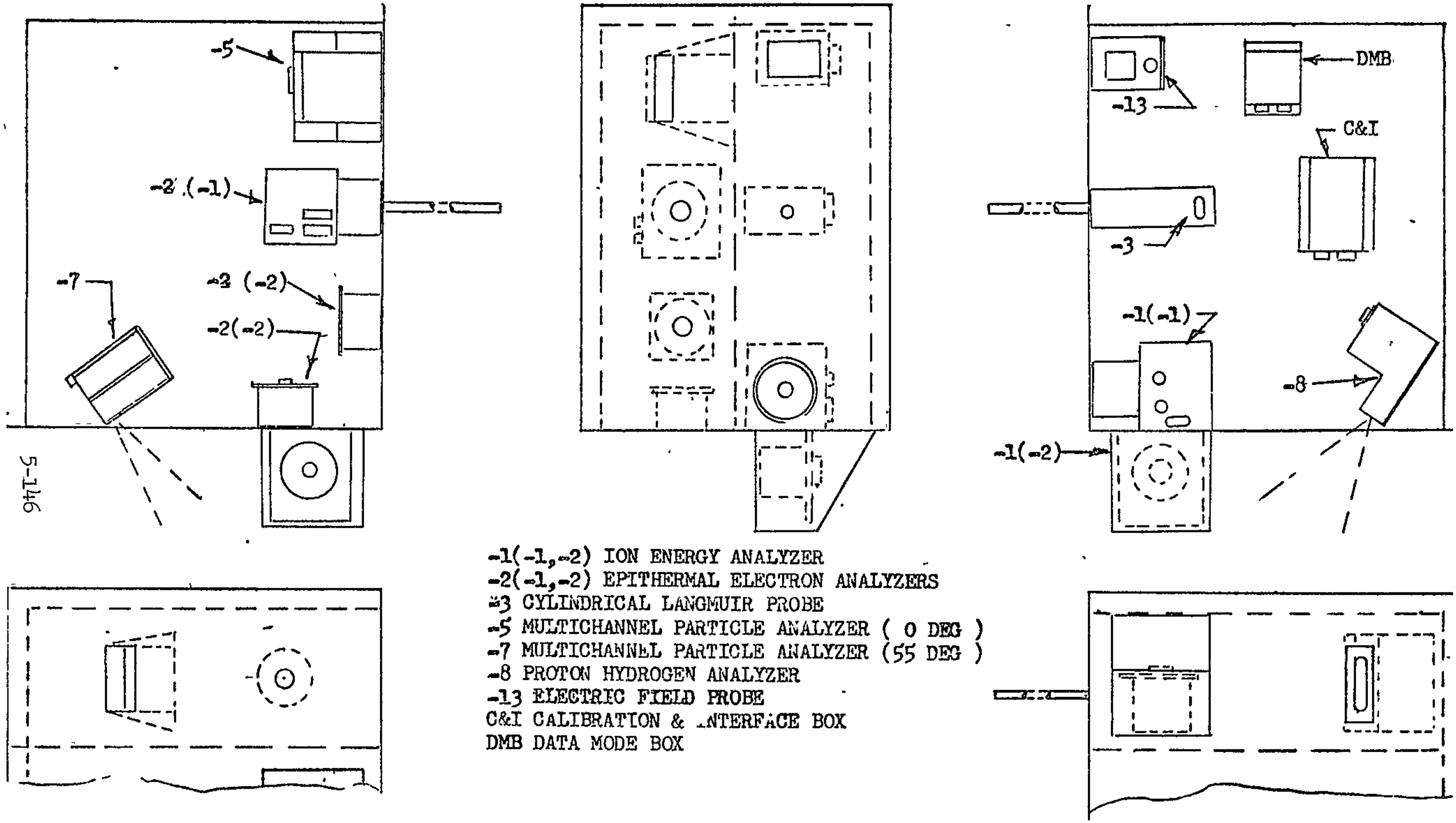
Fig. 5-59b Low-Cost SRS Modules (Shuttle Launched) (2 of 2)

Because of the limited scope of effort allocated to the study of the SRS, no attempt has been made to redesign the experiment equipment. The equipment has been installed into the low-cost SRS in two readily removable modules shown in Figs. 5-60 and 5-61. These spacious modules could accommodate additional experiment equipment. In addition there are two unused compartments in the low-cost SRS which could be used for the installation of equipment to perform other experiments compatible with the HiGlo experiment. More detail concerning the HiGlo experiment may be found in LMSC Engineering Memo PE-41.

5.4.4.4 Stabilization and Control Subsystem. The low-cost SRS is a spin-stabilized vehicle and the Stabilization and Control subsystem has the following functions:

- damp nutation motion induced by separation from the launch vehicle, spinup and despin, and the burning of the propulsion rocket.
- sense the orientation of the vehicle spin axis in space.
- maintain the spin axis normal to the orbit plane which rotates at one degree per day in inertial space.

The Stabilization and Control Subsystem block diagram is shown in Fig. 5-62. After separation from the Space Shuttle (or Expendable Booster) the SRS is spun-up to a nominal 60 RPM by the firing of the spin rockets to provide stability of the propulsion rocket thrust vector. After injection of the SRS into the mission orbit, it is despun to a nominal 10 RPM by the de-spin rockets as required for the operation of the experiment sensors. Nutation damping is obtained from liquid mercury in a toroidal tube. Attitude sensing for readout via telemetry is provided by a pair of IR earth sensors and a digital solar aspect sensor. Controlled spin axis precession is provided by an electromagnetic torquer energized by command in response to roll attitude error signals derived from the earth sensors. The torquer coil consists of 100 turns of No. 23 aluminum wire formed into a coil of approximately $\frac{1}{2}$ in.^{2} (3.2 cm^2) cross section enclosing the spacecraft at the mid-plane. The nutation damper is



-1(-1,-2) ION ENERGY ANALYZER
-2(-1,-2) EPITHERMAL ELECTRON ANALYZERS
-3 CYLINDRICAL LANGMUIR PROBE
-5 MULTICHANNEL PARTICLE ANALYZER (0 DEG)
-7 MULTICHANNEL PARTICLE ANALYZER (55 DEG)
-8 PROTON HYDROGEN ANALYZER
-13 ELECTRIC FIELD PROBE
C&I CALIBRATION & INTERFACE BOX
DMB DATA MODE BOX

Fig. 5-60 Experiment Module No. 1

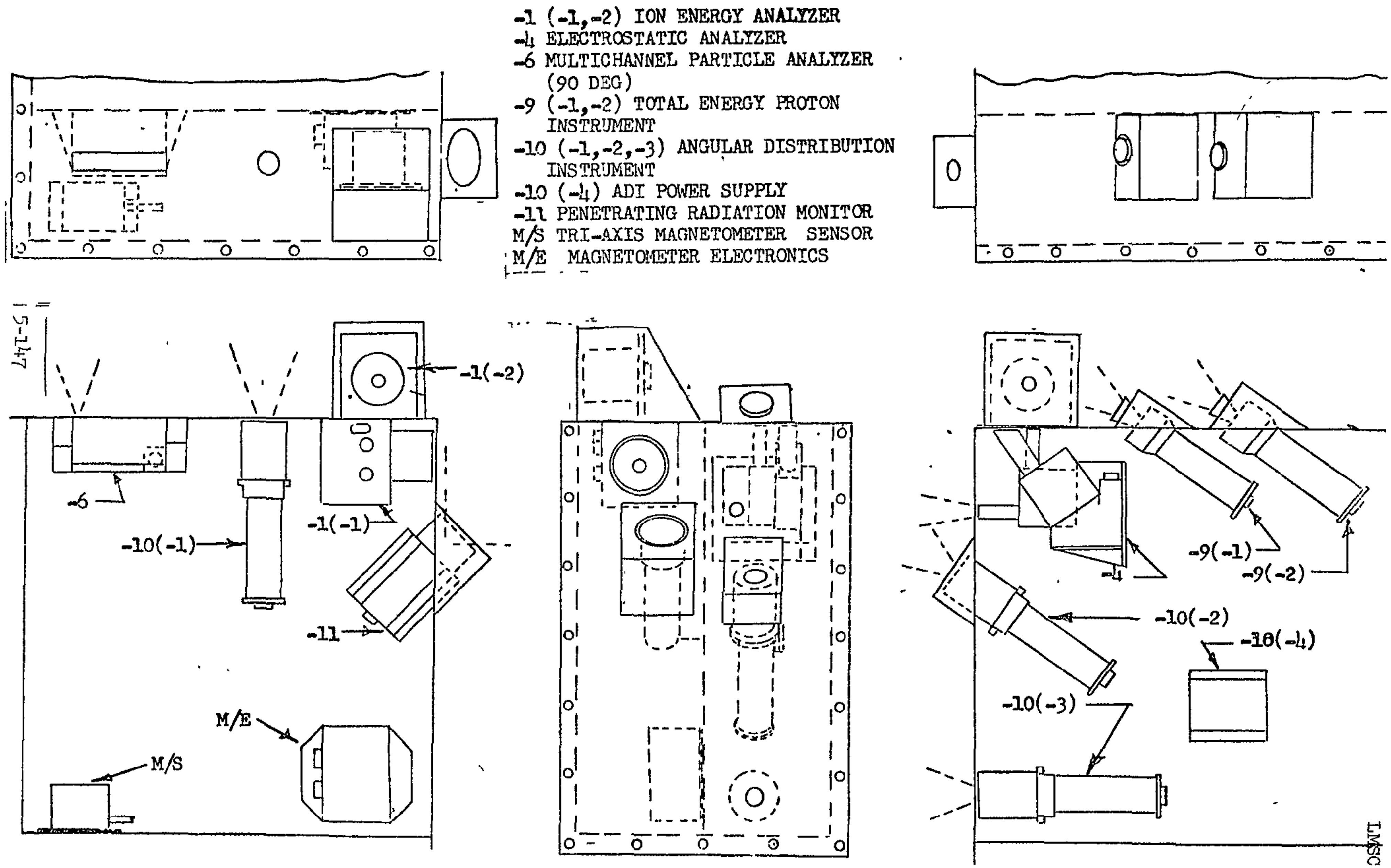


Fig. 5-61 Experiment Module No. 2

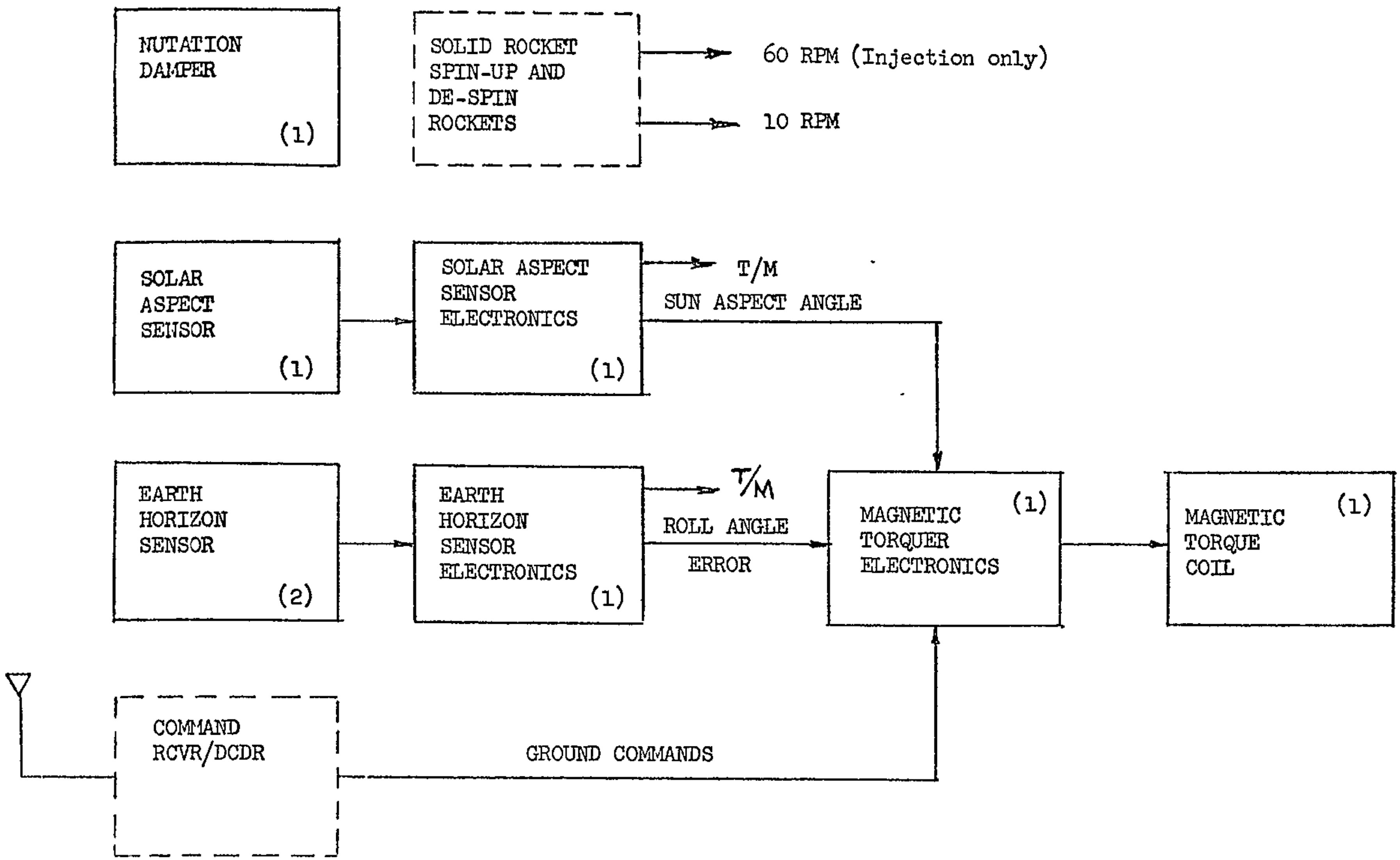


Fig. 5-62 Low-Cost SRS Stabilization & Control Subsystem

attached directly to the spacecraft structure and the remaining S&C subsystem equipment is installed in a readily-removable module. Additional details of the S&C subsystem and supporting analyses are contained in LMSC Engineering Memo, PE-42.

5.4.4.5 Communications, Data Processing and Instrumentation Subsystem. The selected CDPI subsystem design is shown in the block diagram of Fig. 5-63. Experiment data is sampled, and multiplexed for recording, by the 90-point main commutator operating at a frame rate of 5 frames/sec and a 90-point sub-commutator operating at 1 frame/sec. The output PAM data rate is 450 sps and is direct recorded onto one channel of the tape recorder. Recording time up to 120 minutes is available to store experiment data between ground station contacts. Recorder playback is at a 6.3 kbps rate to readout a full data load in 7.5 minutes. The recorded data is linearly combined with the realtime experiment data prior to modulation of the transmitter. The real time experiment data is multiplexed in a commutator operating at 10 frames/sec by using super commutation for the 4 channels to provide a 60 sps sampling rate. The commutator output rate is 300 sps and is FM modulated onto IRIG subcarrier channel 14 by the VCO. The combined baseband output of the tape recorder and VCO FM-modulates a 1-watt transmitter for transmission via a 136-137 MHz dipole antenna.

Spacecraft status data is multiplexed by the 60-point Status Commutator operating at a frame rate of 5 frames/sec. The PAM output is at a rate of 300 sps and is FM modulated onto an IRIG CH 14 VCO. The Earth Sensor and Solar Aspect sensor outputs are FM modulated onto IRIG CH 12 and 13, respectively. All three VCO outputs are linearly summed to provide a composite baseband to modulate the 0.5-watt transmitter. Phase modulation is used and a low modulation index (~1.0 radian, RMS) is selected to preserve a carrier component in the modulation spectrum for tracking by the ground station receivers. The transmitter output is isolated from the command receiver by means of the diplexer and allows use of a common dipole antenna for status telemetry transmission and command reception.

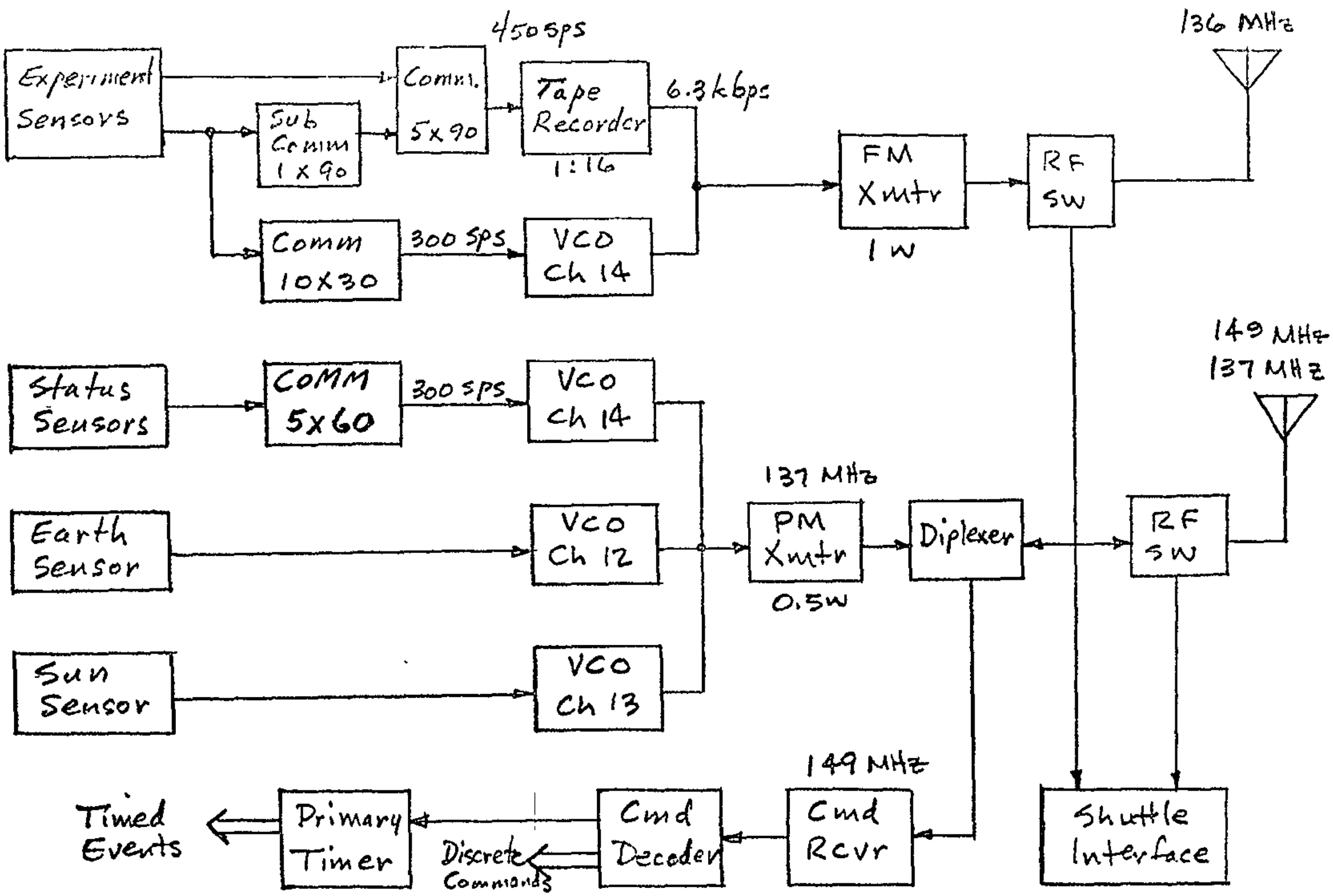


Fig. 5-63 Low-Cost SRS CDPI Subsystem Block Diagram

The command receiver operates at 149-150 MHz and is compatible with the STADAN tone/digital command system. The digital commands consist of 22 bits with the following allocation:

Sync	2 bits
Satellite address	5
Command address	3
Function bits	11
Parity	1
Total	22 bits

The command decoder performs validity checks on the incoming command before execution. These checks will include address, message length, and parity check Command outputs are in the form of relay drivers for discrete commands and digital signals for updating the Primary Timer. The Primary Timer provides 6 pre-programmed events and 8 Orbit-Programmable events.

RF switches between the transmitters and antennas allow hardline checkout of the CDPI Subsystem by the Shuttle checkout console prior to launch.

All the equipment of the CDPI subsystem, with the exception of the two VHF antennas, is installed in a single readily-removable module to minimize cost during development and testing of the subsystem. Additional detail concerning the CDPI subsystem may be found in LMSC Engineering Memo PE-43.

5.4.4.6 Electrical Power Subsystem. The Electrical Power Subsystem of the low cost SRS as configured for the HiGlo mission provides an average of 50 watts of power to the spacecraft subsystems and the experiment equipment. This is sufficient power to sustain operation of experiment equipment approximately 50 percent of the time. The average power output of the Electric Power subsystem may easily be increased to approximately 100 watts by the installation of more solar cell panels in place of blank skin panels and the installation of a second battery, power control unit and battery dissipator. This provision for growth of the Electric Power subsystem makes possible the augmentation of the HiGlo mission or the addition of other compatible experiments to the SRS.

The major equipments for the SRS Electrical Power Subsystem are shown in Fig. 5-64. A silicon solar cell array and a nickel cadmium battery supply continuous power to the system loads during orbital light and dark periods, respectively. The power system provides unregulated dc power to the bus supplying equipment requirements. Using subsystems and equipment condition this dc power as required with individual power supplies. The nominal operating unregulated bus voltage range is 24.5 to 26.5 volts with a maximum range of 22.5 to 29.5 volts.

The solar cell array converts solar energy to electrical energy. The array consists of 8 panels, one on each of the eight sides of the hexagon shaped SRS. The solar cells are phosphorous diffused N on P silicon (2 x 2 cm), 12 mils (0.3 mm) thick with 20 mil (0.51 mm) fused silica coverglasses.

The panel planes are all parallel with the SRS spin axis so that each panel rotates from normal to the sun around to the dark side of the satellite and back to normal to sun while the SRS spins at 10 RPM. The nominal operating temperature of the cells in the sunlight will be approximately 65°F (291°K) and the panel voltage will be 33 volts. The array operating voltage range over a complete orbit is approximately 0 to 50 vdc.

The NiCd battery supplies power to the spacecraft during dark periods and aids in supplying peak loads during light periods. The battery supplies power at 22.5 to 29.5 volts.

The power control unit (PCU) controls input and output of the battery to levels appropriate for the temperature and voltage of the battery.

The battery dissipator provides for discharging the battery through a set of load resistors to periodically recondition the battery by real time command.

The pyro programmer houses the control relays for actuating the pyro devices on the spacecraft, and provides a safe/arm receptacle for pyro circuits.

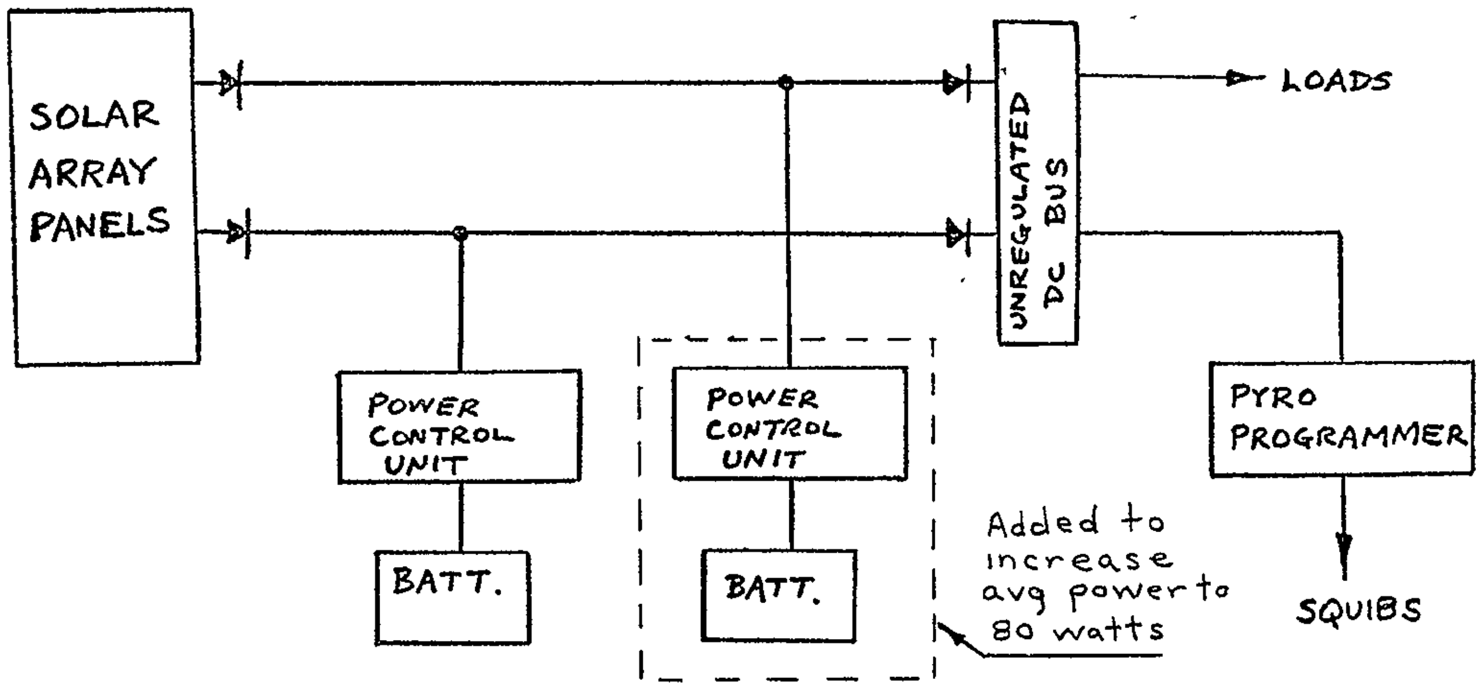


Fig. 5-64 Low-Cost SRS Electrical Power Supply Block Diagram

All of the equipment of the Electrical Power subsystem, with the exception of the solar cell panels is installed in one readily-removable module to minimize development and testing costs.

The solar cell panels are designed to minimize cost. They are all identical to simplify manufacturing. Also the substrate is low-cost aluminum sheet stiffened with standard aluminum extrusion. The cost of solar cells is reduced by the use of the lower efficiency cells from a production lot.

More data concerning the Electrical Power subsystem is contained in LMSC Engineering Memo PE-44.

5.4.4.7 <u>Propulsion and Attitude Control Subsystem</u>. The SRS Propulsion and Attitude Control subsystem performs two basic functions: spin/despin stabilization of the spacecraft, and propulsion of the vehicle into the mission orbit. Spin/despin stabilization is accomplished by firing tangentially mounted solid propellant rocket motors. For the HiGlo mission, orbit adjustment is accomplished by firing a single solid propellant rocket motor mounted on the center line of the spacecraft. Provision is made for the installation of a second rocket motor if additional propulsion is required for an alternate mission.

Each solid propellant spin motor weighs approximately 1 lb (0.454 kg) when fully loaded with propellant and produces 83 lb-sec (369.2 Ns) total impulse. Total impulse may be reduced incrementally to 50 lb-sec (222.4 Ns) by off-loading rocket motor propellant prior to launch thereby compensating for changes in vehicle weight, moment of inertia and spin rate.

To adjust the HiGlo mission orbit, a single solid propellant rocket motor is mounted with the thrust axis directly along the spin axis of the spacecraft. This motor weighs approximately 81 lbs (36.7 kg) when fully loaded. It produces a total impulse between 3,000 lb-sec (13344 Ns) and 17,000 lb-sec (76616 Ns) depending on how much propellant is off-loaded. Therefore, depending on the mission orbit required and the weight of the vehicle, motor performance may be tailored by off-loading propellant. Off-loading is accomplished by the

rocket manufacturer by burning propellant until only the required total impulse remains.

The wide range of impulse available from the spin motors and the primary propulsion motor provides for growth of the spacecraft mass and inertia and for variations in mission orbit requirements. Additional data concerning the Propulsion and Attitude Control subsystem is contained in LMSC Engineering Memo PE-45.

5.4.4.8 Environmental Control Subsystem. Because of the attitude of the SRS in orbit and the large surface area of the SRS relative to the quantity of heat dissipated in the internal equipment, the bulk average temperature of the spacecraft will be relatively low, probably between $0^{\circ}F$ (255°K) and 30°F (272°K). Heaters and insulation will be required to maintain the battery at the desired temperature of 32°F (273°K). The ends of the spacecraft, which are always parallel to the orbit plane - sun-line, will be insulated with 1 in. (2.54 cm) thick blankets of multilayer insulation. All interior surfaces will be bright aluminum and all plain skin panels will be finished externally to have high α/ϵ surfaces. Additional discussion of the thermal control requirements of the SRS is contained in LMSC Engineering Memo PE-46.

5.4.4.9 Summary of Weights for Low-Cost SRS. The Shuttle-launched SRS weight summary by subsystem is shown in Fig. 5-65. The total weight, including 15 percent contingency, is 697 lb (316.2 kg) and compares with the weight of the Corrected Baseline SRS, 317 lb (143.8 kg). Figures 5-55a through 5-55f, shown previously, show the detailed weight breakdown of four configurations of the SRS: the Initial Baseline, the Corrected Baseline, the Low-Cost (Expendable Booster) and the Low-Cost (Shuttle Booster).

5.4.4.10 Reliability. A comparison of the reliability predictions for the Corrected Baseline SRS and the Low-Cost SRS (Shuttle-launched) is shown in Fig. 5-66. They are seen to be nearly equal as required by the study ground rules. However, further analysis may result in lowering the reliability requirements for the Shuttle-launched SRS when consideration is given to the abort and in-orbit checkout capability of the Shuttle.

Subsystem	Corrected Baseline SRS (lbs)	Low-Cost SRS Without Contingency (lbs)	*Low-Cost SRS With Contingency (lbs)
Experiment Installation	55.7	82.3	94.6
Electrical Power Subsystem	99.8	144.9	166.6
Communication, Data Processing & Instrumentation Subsystem	61.4	57.4	66.0
Stabilization & Control Subsystem	26.3	53.3	61.3
Propulsion & Attitude Control Subsystem	27.0	48.0	55.2
Structure & Mechanisms Subsystem	40.3	180.2	207.3
Environmental Control Subsystem	6.4	40.0	46.0
Total Payload Weight	316.9	606.1	697.0

* Including weight contingency of approx. 15%.

1 lb = 0.4536 kg

Fig. 5-65 Weight Summary - Low-Cost SRS - Shuttle-Launched

Subsystem	Corrected Baseline SRS	Low-Cost SRS Shuttle-Launched
Structure & Mechanisms	.9985	.9985
Environmental Control	.9994	.9994
Communications, Data Processing & Instrumentation	.952	.8777
Electrical Power	.9077	.9381
Stabilization & Control	.987	.9532
Propulsion & Attitude Control	.9139	.9139
Experiment Installation	.8247	.8247
Payload Total	.6408	.5903

Fig. 5-66 Comparison of SRS Reliability

5.4.5 Expendable-Launched SRS Performance and Design Requirements

The low-cost expendable-launched SRS is designed to be launched by expendable launch vehicles: the Atlas/Burner II or the 3-Segment SRM/Titan Core II/Agena.

The basic mission operations plan for the use of the low-cost expendable-launched SRS in the performance of the HiGlo experiment mission requires that two vehicles be launched from a single booster. The designated boosters provide much excess payload capability [approximately 2500 lbs (1134 kg)] for this mission, and could place 4 vehicles into orbit. Two of these could be configured for another mission.

5.4.6 Low-Cost Expendable-Launched SRS Configuration

The general configuration of the low-cost expendable-launched SRS is the same as that of the low-cost Shuttle-launched SRS. The expendable-launched SRS is derived from the Shuttle-launched SRS by the addition and deletion of equipment as shown in Fig. 5-67. Most of the subsystem equipment is installed in readily-removable modules as detailed in Figs. 5-68a, 5-68b, and 5-68c.

5.4.7 Description of Low-Cost SRS (Expendable-Launched)

5.4.7.1 Subsystems of the SRS. The descriptions of the Shuttle-launched SRS subsystems are, in general, applicable to the expendable-launched SRS subsystems and will not be repeated. Only the differences between the subsystems of the expendable-launched SRS and those of the Shuttle-launched SRS will be described.

5.4.7.2 Stabilization and Control Subsystem (S&C). The subsystem is augmented by an Auxiliary Flight Control Electronics unit to provide backup to the primary attitude sensing equipment and to increase confidence in the success of the mission; since the confidence provided by in-orbit checkout prior to launch afforded by the Shuttle is not available. The Auxiliary Flight Control Electronics unit is described in more detail in LMSC Engineering Memorandum PE-42.

ADDED	DELETED
Electrical Power Subsystem	
1 NiCd Battery, 12 AH	
1 Battery Dissipator	
1 Power Control Unit	
Communication, Data Processing, & Instrumentation Subsystem	
1 Command Receiver	2 RF Switches
1 Tape Recorder	
Stabilization & Control Subsystem	
1 Auxiliary Flight Control Electronics	

Fig. 5-67 Equipment Changes to Shuttle-Launched SRS for Expendable-Launched SRS

Subsystem	Module	Equipment in Module	Module Weight (lb)
Experiment Installation	Experiment No. 1	Ion Energy Analyzer (-1) Ion Energy Analyzer (-2) Epithermal Electron Analyzer (-1) Epithermal Electron Analyzer (-2), 2 Cylindrical Langmuir Probe 0° Multichannel Particle Analyzer 55° Multichannel Particle Analyzer Proton Hydrogen Analyzer Electric Field Probe Calibration & Interface Box Data Mode Box Module Base Module Cables	Basic 37.4 15% Cont. 5.6 Total 43.0
Experiment Installation	Experiment No. 2	Ion Energy Analyzer (-1) Ion Energy Analyzer (-2) Electrostatic Analyzer 90° Multichannel Particle Analyzer Total Energy Proton Sensor (-1) Total Energy Proton Sensor (-2) Angular Disttribution Instrument, 3 ADI Power Supply Penetrating Radiation Monitor Tri-Axis Magnetometer TAM Electronics Module Base Module Cables	Basic 44.9 15% Cont. 6.7 Total 51.6

1 lb = 0.4536 kg

Fig. 5-68a Low-Cost SRS Modules (Expendable-Launched) (1 of 3)

Subsystem	Module	Equipment in Module	Module Weight (lb)
Electrical	Electrical Power	NiCad Battery, 12 AH, 2 Battery Dissipator, 2 Power Control Unit, 2 Pyro Control Unit Module Base Module Cables	Basic 91.6 15% Cont. 13.7 Total 105.3
Communication, Data Process., & Instrumen.	Communication & Data Processing	Command Receiver, 2 Command Decoder Primary Timer Tape Recorder, 2 Status Commutator (5x90) Experiment Commutator (5x90) Experiment Commutator (1x90) Experiment Commutator (10x30) Voltage Controlled Oscillators, 4 FM Transmitter FM Transmitter Diplexer Module Base Module Cables	Basic 66.0 15% 9.9 Total 75.9

Fig. 5-68b Low-Cost SRS Modules (Expendable-Launched) (2 of 3)

Subsystem	Module	Equipment in Module	Module Weight (lb)	
Stabilization & Control	Stabilization & Control	Earth Horizon Sensor, 2 Earth Horizon Sensor Electronics Solar Aspect Sensor SAS Electronics Magnetic Torquer Electronics Auxiliary Flight Control Electronics Module Base Module Cables	Basic 15% Cont. Total	29.3 4.4 33.7

Note: Following equipment is not part of any module and is mounted directly on Spacecraft structure

Solar Panel (8 or 13)
Spacecraft Electrical Harness
VHF Antenna (2)
Wobble Damper
Magnetic Torquer Coil
Spin Motor (4)
Despin Motor (4)
Primary Rocket Motor
Thermal Insulation and Coatings

Fig. 5-68c Low-Cost SRS Modules (Expendable-Launched) (3 of 3)

5.4.7.3 Communication, Data Processing, and Instrumentation Subsystem (CDPI). The subsystem is augmented by the addition of a second command receiver and a second tape recorder to provide redundancy for the vital functions of command reception and data recording, and to increase confidence in the success of the mission. Precedent for these redundancies was established by the corrected-Baseline SRS.

The RF switches included in the CDPI subsystem of the Shuttle-launched SRS have been deleted since their function, to provide for checkout in the Shuttle cargo bay, is not applicable.

5.4.7.4 Electrical Power Subsystem. The subsystem is augmented by the addition of a second battery, power control unit, and battery dissipator to provide redundancy of energy storage equipment, as in the Corrected Baseline SRS, and greater confidence in mission success.

5.4.7.5 Weight Summary of the Low-Cost SRS (Expendable-Launched). The weight of the low-cost expendable-launched SRS, by subsystem, is shown in Fig. 5-69. Detailed weights of the Initial Baseline SRS, the Corrected Baseline SRS, the Low-Cost Expendable-Launched SRS, and the Low-Cost Shuttle-Launched SRS may be found in Figs. 5-55a through 5-55f.

5.4.7.6 Reliability. Predicted reliability of the low-cost expendable-launched SRS subsystems is shown in Fig. 5-70. The predicted reliability of the corrected baseline SRS subsystems is shown for comparison.

Subsystem	Corrected Baseline SRS (lbs)	Low-Cost SRS Without Contingency (lbs)	*Low-Cost SRS With Contingency (lbs)
Experiment Installation	55.7	82.3	94.6
Electrical Power Subsystem	99.8	182.9	210.3
Communication, Data Processing, & Inst. Subsystem	61.4	67.5	77.6
Stabilization & Control Subsystem	26.3	55.3	63.6
Propulsion & Attitude Control Subsystem	27.0	48.0	55.2
Structure & Mechanisms Subsystem	40.3	180.2	207.3
Environmental Control Subsystem	6.4	40.0	46.0
Total Payload Weight	316.9	656.2	754.6
* Including weight contingency of approximately 15%			

1 lb = 0.4536 kg

Fig. 5-69 Weight Summary - Low-Cost SRS - Expendable Launched

Subsystem	Corrected Baseline SRS	Low-Cost SRS Expendable-Launched
Structure & Mechanisms	.9985	.9985
Environmental Control	.9994	.9994
Communication, Data Processing & Instrumentation	.952	.9250
Electrical Power	.9077	.9570
Stabilization & Control	.986	.9860
Propulsion & Attitude Control	.9139	.9139
Experiment Installation	.8247	.8247
Payload Total	.6408	.6565

Fig. 5-70 Subsystem Reliability - Expendable-Launched SRS

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Section 6
PROGRAM PLANS AND COSTS FOR LOW-COST PAYLOAD SYSTEMS

This section summarizes the program planning and costing approach used during the Payload Effects Analysis Study to estimate costs of the low cost designs and documents the resulting plans and costs.

The payloads selected for analysis were the OAO-B, SEO, MO and SRS. Following the low-cost redesign of OAO-B, the planning and cost estimating was completed. Review of the results by NASA indicated the desirability of examining the baseline costs in more detail. It was agreed that the baseline OAO-B and SEO would be recosted using the same basis as the low-cost OAO-B and SEO were costed. This was accomplished following the planning and costing of the SEO. Time limitations coupled with this new recosting task led to the elimination of the MO from consideration. Following the baseline recosting, the low-cost SRS plan and costs were developed. Changes to the baseline SRS design resulted in adjustment of the baseline costs; however, the baseline was not recosted as there were sufficient data available for comparison of SRS configurations.

The groundrules and assumptions used for developing the program plans and costs are summarized for OAO-B, SEO and SRS, and the planning, estimating and costing approaches are described. The development hardware is described and summarized by payload program. As testing is a major cost contributor, the overall testing approach is discussed and the general requirements for payload support equipment are summarized. A general summary of payload cost reduction techniques found during the study is provided.

The development plans for each of the payloads, groundrules for recosting the baselines and program costs are summarized. The costs for each payload and for each launch configuration are compared and, finally, the results are analyzed and the cost impact of the advanced transportation systems is examined.

Following accumulation of the costs of the payloads and their subsystems, the specific cost-impacting payload effects were analyzed. Cost drivers such as mass and volume effects, modular construction, use of simpler components and lowered reliability requirements were identified and quantified. Unit payload cost was shown to have a sizable impact on the cost of development and qualification test hardware which in turn impacted overall RDT&E costs. For example, on OAO-B, a total program savings of 16 percent in unit cost contributed to an R&D hardware and test savings of 57 percent. Similar savings were apparent in SEO and SRS. Savings afforded by Shuttle when compared to LCE were also identified in a similar manner.

Application of 1970 technology had a major effect upon the costs of the OAO-B amounting to 16.8 percent of RDT&E cost savings and 20 percent of unit cost savings. This effect was separated from the OAO and the results compared with SEO with excellent correlation. The results indicated that, without applying new technology, savings of about 30 percent were possible due to the other payload effects afforded by Space Shuttle, exclusive of refurbishment.

An analysis of the overall impact of payload refurbishment upon total payload program costs indicated that the capability offered by Space Shuttle/Space Tug to retrieve payloads from orbit for refurbishment was the major savings offered by introduction of new space transportation systems. This analysis, which compared the costs of Shuttle-launched and LCE-launched payloads, normalized such payload effects as weight and volume and 1970 technology. The results showed an approximate savings of 50 percent in the cost of space programs (51 percent on OAO and 50 percent on SEO).

6.1 GROUNDRULES AND ASSUMPTIONS

The groundrules and assumptions used in developing the program plans and in estimating costs are summarized. The general approach used on OAO-B and SEO are consistent with current NASA/DOD program guidelines. To maintain similarity to the overall P-11 approach, i.e., low-cost and quick response, variation in groundrules was introduced for the SRS. These changes are also described.

6.1.1 OAO-B

The following assumptions and guidelines were used in developing the plans and costs for the low-cost OAO-B:

- Flight objectives and general development approach for the OAO-B are maintained constant.
- NASA Phased Project Planning Guidelines (NHB 7121.2) apply. The planned program starts with initiation of Phase B. There will be a six-month evaluation, proposal and award period between Phases B and C, and a three-month period between Phases C and D.
- The operational orbital lifetime is one year.
- The basis for detailed design, plans and cost is the Space Shuttle-launched configuration. Differences in costs are summarized for the alternate expendable launch systems, the Titan TIII-L2 and the SLV-3C/Centaur.
- A prime aerospace contractor will be selected by NASA to conduct the program under direction of a NASA program office. NASA will maintain overall project management responsibility. Prime contractor fee is not included in overall costs.

- Present NASA/DOD and industry approaches to aerospace program management will be followed.

- Plans, estimating approaches and cost breakdowns should be comparable to the baseline OAO program.

- GFE facilities and services for OAO include launch vehicles, shrouds, shuttle adapters, NASA launch services, the GSFC OAO Control Center, STADAN facilities, NASCOM communications network and the GSFC operational computer systems. These are no-cost items.

- Advantage is taken of 1970 state-of-the-art. In order to provide comparability, a total program from concept through flight has been planned. Use of OAO unique designs was avoided unless such designs are now generally available to the industry.

- All new developments undergo qualification testing. Qualified off-the-shelf components developed for other space program applications will not require certification. Limited development testing of the new designs is planned. Reliability testing is included for all configurations.

- A single flight payload will be delivered.

- Cost of module/spacecraft refurbishment is not included.

- A qualification test vehicle (QTV) will be assembled and used during the development phases. For the shuttle program, qualification testing culminates in a sortie flight of the QTV. Costs for the sortie flight are included in the RDT&E phase. The QTV will be made as a flight equivalent of the Flight Test Vehicle (FTV) and can serve as a backup FTV and as a source of spares for orbital maintenance and refurbishment. Shuttle sortie costs are $3 million.

- Two sets of standard automatic checkout equipment, separately developed, including standardized software, will be available for ground and shuttle use. Program peculiar software costs are borne by the program. A use charge of $2.1 million each for two sets of equipment and associated common software is charged to the program.

- The contractor has complete test facilities available for the low-cost OAO-B/Shuttle program. Costs included are equivalent to government costs for providing NASA facilities as GFE.

- Planning, management and reporting are consistent with current NASA practice for unmanned R&D space missions. Essential quality assurance inspection or test has been included. The provisions of NHB 5300.4 (1 A&B) have been adhered to.

6.1.2 SEO

The assumptions and guidelines used in developing the plans and costs for the low-cost OAO-B pertain to SEO plans and costs except as noted:

- The development approach for the SEO is maintained, where possible, consistent with Lunar Orbiter.

- The planned SEO program starts with initiation of Phase B. There will be a six-month evaluation, proposal and award period between Phases B and C, and a combined Phase C/D.

- The operational orbital lifetime is two years nominally. Program duration is greater than two years.

- The basis for detailed design, plans and cost is the Space Shuttle/Space Tug launched configuration. Differences in costs are summarized for the alternate expendable launch system, the Titan IIID/Centaur.

- Plans, estimating approaches and cost breakdowns should be comparable to the baseline SEO program and to the Lunar Orbiter.

- Five flight payloads and two sets of non-redundant spare modules will be delivered. The fifth payload will serve as backup to No. 4 and as the initial replacement spacecraft.

- As with Lunar Orbiter, a subcontractor will develop the photo payload; subcontractor costs have been estimated. A 7 percent subcontractor fee was included.

- A Development Test Vehicle (DTV) will be assembled and used during the development phase. The DTV will be modified and updated to flight configuration for qualification testing, training and operational readiness demonstration (ORD) as a Qualification Test Vehicle (QTV). A shuttle-sortie test flight is not required for SEO.

- Each initial launch spacecraft will be backed up by the next spacecraft which shall be in flight ready status.

- Four sets of standard automatic checkout equipment, separately developed, including standardized software, will be available for ground and Shuttle use. Program peculiar software costs are borne by the program. A use charge of $1.6 million each for four sets of equipment and associated common software is charged to the program.

6.1.3 <u>SRS</u>

The following assumptions and groundrules were used in developing the SRS program plans and costs:

- The objectives of the ARPA HIGLO experiment are retained. A minimum of two years' continuous orbital observation is planned.

- Four flight spacecraft will be delivered. Nominal individual payload lifetime is estimated at 6 months.

- The basic approaches used in developing and operating the P-11 subsatellite will be followed. The program is experimental and of a quick-response nature.

- A prime aerospace contractor will be selected to conduct the program under the overall direction of a government program office. Close government/contractor liaison by direct government participation at the contractor's facility is assumed.

- The span from go-ahead to first deployment will be 2 years.

- NASA-phased Project Planning Guidelines are not generally applied. Program will start with award. A conceptual design review will be held at earliest practicable date. Procurement of long-lead materials and components will be initiated following approval of the initial program (Part I) specifications and the advanced material requirements (AMR) documents.

- A hard mockup will be built as an engineering tool and aid to manufacturing planning and tool design.

- There will be no structural test article (STA), development test vehicle (DTV) nor qualification test vehicle (QTV) for Shuttle application. The first production vehicle will be used for qualification and reliability testing. This vehicle will be reworked to serve as the fourth flight spacecraft. The second production vehicle (FTV) will be the first delivered for launch. For the ACE/LCE, a QTV will be developed and can serve as flight backup.

- All flight spacecraft will undergo comprehensive acceptance testing including magnetic cleanliness, anechoic, thermal-vacuum and acoustic

tests. All flight spacecraft will be spin-balanced with dummy pyrotechnics installed. Installation of ordnance units will be accomplished at the launch base.

- Flight experiment packages will be delivered prior to completion of final assembly of individual spacecraft.

- All essential quality assurance testing and inspection will be accomplished.

- Formal documentation will be minimized and deliverable documentation is summarized below:

  Monthly technical progress and financial management reports in one package

  Part I and Part II (Hardware, Software and Interface) Specifications

  Test procedures, reports and summaries

  Acceptance Data Package (As-built drawings and supporting design data)

  Interface and Compatibility Test Specifications, Procedure and Report

  Launch and Flight Operations Procedures

- Drawing release and configuration management procedures will be consistent with LMSC Procedure E-229 dtd 5-14-70 which has been successfully applied to the P-11 Program.

- Refurbishment is not required and costs are not included.

- The following equipment, services and facilities are to be government-furnished and costs are not shown in the SRS program cost summaries:

Government program office expense

Launch Vehicles - Shuttle, LCS or ACE

Expendable launch vehicle shrouds

Shuttle/Payload Adapter and Interface Equipment

Shuttle On-Board Checkout Equipment; one set of equipment will be provided for use at the Contractor's facility for final acceptance testing and for interface verification

Launch and Mission Operations Services

Launch facilities

Tracking & Data Acquisition, Mission Control and Network communications

Experiment Package

- A minimum sized program office will be maintained for 2 years following launch to provide support to the government for mission operations.

- Learning curve will not be applied to unit cost. First unit cost = Average Unit Cost.

- Payload Effects have not been applied to the experiments. Costs for experiments included in the summary are the same as the baseline.

6.2 GENERAL PLANNING AND COSTING APPROACH

Based upon groundrules and assumptions summarized in Section 6.1, an overall program plan from program initiation through launch, was developed for each design. Design and performance specifications, drawings, parts lists, reliability estimates and weight summaries were provided by the low-cost design activity. "Bottom-up" cost estimates were completed. Resulting costs are considered to be typical for the general aerospace industry using 1970 dollars and rates. The general approach used consisted of analyzing the design data, defining elements of work to be accomplished, estimating labor and material required, scheduling work to be accomplished and applying typical dollar rates

to determine final cost estimates, including direct and burden costs. These costs, in turn, were spread by subsystem and compared with the previous parametric cost targets. Costs were also compared by ratio with current LMSC programs of equivalent magnitude and complexity to evaluate the results. Finally, the results were audited by financial analysts and reviewed by management.

Significant cost drivers affecting each design planned and costed include the need for development hardware, the test planning approach employed, and the assumptions concerning ground support equipment. Included in this section is a general discussion of the assumptions utilized in this Study concerning each of these cost affecting parameters. Also the costing and estimating approach utilized is described.

6.2.1 General Planning Information for Development Hardware

The equipment used on the low-cost payloads is a mixture of new developments and off-the-shelf equipment. Depending on the item's design maturity and application to the mission, specific developments will be required to achieve flight status. During the development process, the new-development items will evolve from breadboards for new developments, through engineering Development Test Units (DTU) to flight-type prototypes which are highly representative of anticipated deliverable equipment. Although individual items may start at any point in the development process, they will all achieve prototype status before assembly into a Qualification Test Vehicle (QTV).

Quality verification is applicable to prototype equipments and includes these product assurance and quality control functions that are to be used for deliverable equipments. However, during development, the resolution of quality infractions is a function of engineering judgment rather than formal review procedures. During design development, prior to such quality verification, engineering disclosure for breadboards and DTUs are locally controlled by engineering through a release and change control system. Upon design approval, the engineering disclosure is brought up-to-date and released into a rapid-response

control system. The DTUs are then upgraded to prototype status and quality verification performed. Prototype equipments are used for design certification.

The conservative design approach permits early commitment to final packaging because subsequent design changes resulting from development testing can be implemented without significant changes in the equipment or without destroying the validity of previous testing. Because of this approach, evolutionary upgrading is practical without significant scrapping of developing hardware, and the QTV will be equivalent to the flight-type vehicle (FTV). Throughout the development program, operations conducted and procedures used will be similar to the activities planned for use on deliverable equipment to provide experience and training prior to exposing deliverable equipment to potentially damaging conditions.

Upon completion of the development program, other uses of the QTV or its equipments can be planned. Depending upon the results of development testing, many of the equipment may be usable as spares, and the QTV could be used as a backup to the FTV. Specific determination cannot be made until the amount of testing, rework, and quality of the individual items is known. However, the QTV will represent a significant resource to the overall program in the form of spare parts, etc.

Because SEO approaches a production-type program, a formal qualification program will be used to certify component and system assemblies. Therefore, the set of component DTUs used for development will be specified to assemble the Development Test Vehicle (DTV). After system development the DTV will be upgraded by prototype components into a Qualification Test Vehicle (QTV).

The QTV will be qualification tested through a series of functional and environmental tests representing mission conditions. Subsequent to qualification, the QTV will be used for ground simulation and operation training and for operational readiness demonstrations (ORD). On SRS, the QTV is modified

and updated as the 4th FTV for an additional cost-saving approach. A summary of flight and development hardware for all low-cost programs and for the baseline OAO-B and SEO is provided in Fig. 6-1.

6.2.2 Testing and Test Planning Approach

Development, qualification, reliability and acceptance testing represent between 25 and 40 percent of total program costs. New design concepts and test philosophies affecting testing can contribute materially to program cost reduction. As an example, by relieving weight constraints, structural design factors can be increase beyond the point where structural testing is required. This will directly impact RDT&E costs by reducing test hardware, test equipment, and man-hours. Use of qualified components will eliminate qualification testing at the component level. Thus an inspection of the overall testing approach is merited.

Test programs are required to demonstrate the performance capability of the spacecraft. To effectively implement a test program, the overall program is subdivided by different test objectives related to specific hardware assemblies. Therefore, the test program is composed of development, qualification, reliability and acceptance tests conducted on component, module or subsystem, and system hardware assemblies. The results of the test engineering activity is an integrated plan to accomplish the objectives in a cost effective manner by evaluating the factors affecting the test program such as design maturity, previous experience, analytical capability, and risk. In the final analysis, testing can improve the quality of equipment only as the test results are used to influence the design and production requirements. When confidence is high (or risk is low) based upon experience and program objectives, testing should be reduced.

The Payload Effects on test costs are a result of relaxing design constraints, which permits more conservative designs. As designs become more conservative, more reliance can be placed upon experience to develop confidence in quality

	OAO			SEO			SRS		
	Base	LCE/ ACE	Shuttle	Base	LCE/ ACE	Shuttle	Base	LCE/ ACE	Shuttle
Flight Spacecraft (FTV)	1	1	1	4	4	4	4	4	4
Backup FTV	← Refurb. QTV →			1	1	1	← QTV avail. for refurb. →		0
Qualification Test Vehicle (QTV)	← DTV Structure & Prototype Equip. →			2	1 Plus DTV Re-furb.	DTV Re-furb.	1	1	First FTV used for Qual.
Prototype Components (sets for Qual.)	2	1	1	2	1	1	0	0	0
Development Test Vehicle (DTV)	← Structure & Devel. Equip. →			← Structure & Develop.Equip. →			0	0	0
Development Components (sets for devel.)	1	1	1	1	1	1	← New Devel. only) →		
Struct. Test Articles (STA)	2	0	0	2	0	0	0	0	0

Fig. 6-1 Summary of Flight and Development Test Hardware

and to reduce unanticipated design and production deficiencies. Increasing weight and volume can reduce the criticality of design approaches and production operations, which permits simplified, integrated testing without increasing risk. For Shuttle launched payloads, additional benefits are possible through on-orbit checkout and maintenance, further reducing ground test operations. In all cases, complete system performance is demonstrated prior to operational commitment. The test programs delineated for this study are consistent with current NASA/DOD approaches. Any reductions in test costs are a result of effects generated by conservative designs, which were less sensitive to environment and functional interface variations and more amenable to analytical verification.

A common approach was used for generating the test programs for all spacecraft included in the study. The primary factors causing differences between the low-cost spacecraft test programs were the:

- Degree of sophistication required for equipment performance indicated by tolerance levels and the amount of different operational modes (system complexity),
- Availability of mature equipment designs,
- Total quantity of equipments and spacecraft. (Particularly with regard to facilities and support equipments.)

All programs demonstrated:

- Acceptance performance at the component, module (if applicable), and system assembly levels. Acceptance testing included environmental stressing to detect production deficiencies.
- Qualification of system performance, and separate component qualification tests as appropriate.

- Development performance of new or extensively modified equipments and complex subsystems.

The scheduling of tests was generally consistent with NASA phased project planning in that breadboard feasibility developments were related to preliminary design, product developments and component qualifications preceded hardware go-ahead and system qualification preceded flight vehicle acceptance. Usually, the initial Critical Design Review (CDR) approved component design to permit initiation of production procurement, and final CDR approved system design prior to first flight vehicle assembly. This approach using progressive CDRs permitted optimum phasing of program activities without jeopardizing the phased project approach.

To develop the spacecraft test program, individual equipments were identified and evaluated to determine their degree of sophistication and availability. Where warranted, individual breadboard and product development tests were conducted which included environmental stressing to detect critical weaknesses. Subsystems were assembled and tested by procuring additional non-redundant equipment, which did not require individual development. Except for SRS, Development Test Vehicles (DTV) were eventually assembled using the development equipment and structures to determine system interactions. Concurrent with system development, qualification of prototype components was initiated where previous test history would not suffice. Qualification Test Vehicles (QTV) were used to demonstrate compatibility with support equipment, activate and train system test operations, conduct system qualification tests, and conduct Operational Readiness Demonstrations at the launch site. Acceptance test operations on flight vehicles followed qualification to retain proficiency developed on the QTV. Where multiple deliveries were required, test operations were scheduled continuously to maintain stable work loads and to increase efficiency.

Test hardware is a significant cost element in the RDT&E test program, so the low cost approach was structured to make maximum use of hardware by refurbishment and reuse, and by scheduling series vs concurrent operations. Generally,

only one unit was required for qualification testing. This approach is consistent with the increased confidence attributed to conservative designs.

Experience on past aerospace programs indicates continued improvement in the design and production of quality components. This trend should reduce test costs, but increases in system complexity have offset any gain in capability. A decrease in design constraints should lead to better initial designs, insensitive interfaces between equipments (increased margins), and reduced workmanship deficiencies. As experience demonstrates improved quality, the trend of rising test costs should be reversed.

6.2.3 Ground Support Equipment

Major items of support equipment are required at the operational launch and mission support sites and at the contractor plant. Equipment is needed to test, simulate, service, handle and transport the spacecraft and its modules. For the purpose of cost segregation, support equipment is categorized as GSE, GHE, or STE where GSE and GHE are associated with operational uses and STE is used by the contractor, in-plant. Where commonality exists between GSE & STE, GSE is used in-plant also. Typical GSE includes:

- System test complexes
- Sensor targets (for test simulation)
- Experiment module interface test sets
- Pneumatic servicing equipment
- Photographic servicing equipment for SEO
- Thermal-vacuum heat flux simulators
- Software
- Interconnecting cables

Automatic checkout and test complexes are standardized test stations developed independently and provided for a fixed fee. The standard automatic checkout system provides the capability to interface with the spacecraft by providing electrical power, RF data/command interfaces, programming sequencing and

monitoring functions. Program peculiar modifications may be required to provide recording, real time displays, and timing functions. Standard test software packages provided with the basic system are adapted to the individual spacecraft requirements. Standard test sets are required at the contractor plant at the launch base and installed in the Shuttle for predeployment checkout. Further description is provided in Section 8.2.

GHE includes:

- Spacecraft handling dolly and transporter
- Module handling dollies and installation equipment
- Work stands

Based upon the production quantity and rates, specialized handling equipment may be justified. These will be capable of considerable reuse. Vehicle dollies will be required at the contractor plant and at the launch base. Two transporters were provided to support backup launch operations. Module dollies are required of each configuration to support production at the contractor's plant and spares at the launch base. One installation fixture is provided for each configuration at the plant and launch base. Reusable shipping containers for modules will be provided. These containers will be fabricated of common packaging durable materials and reusable fasteners. Some wearout and scrappage has been traded-off against hard packaging.

STE encompasses the remaining contractor support equipments and fixtures to process and test the spacecraft and its equipments. Typical STE includes:

- Collimators for OAO
- Equipment/module checkout stations
- Spacecraft interface simulators/fixtures
- Experiment module/spacecraft interface simulators
- Weight/balance equipment
- Hoists/slings/rigging equipments
- Environment test fixtures
- Spin balance equipment for SRS

Mission peculiar support equipment, for installation and use in data acquisition, display and processing and for spacecraft control at the mission control center and at the tracking station was specified for each program and were costed.

Other equipment, such as teletypes, intercoms, tape recorders, card and tape punches, central support computers, office furnishings, etc., have been assumed to be existing and available as GFE, along with the necessary rooms and utilities. Full support, including general operation of the control room, communications equipment and ground stations, will be provided by the NASA OTDA.

Pre-launch operations are assumed to be at KSC, with suitable facilities assigned to the SEO project to provide support for operations such as receiving, checkout, launch preparation, integration with Tug and Shuttle, launch, retrieval and repair and maintenance of returned spacecraft. Support of a KSC ground station (DSS-71 or equivalent S-band equipment facility) is assumed.

6.2.4 Costing and Estimating Background

Estimating. Labor estimates for development, production and operation were made, in accordance with the assumptions and guidelines by engineering, manufacturing, test and operations planners. Lists of materials were revised to determine a representative make-or-buy list. Procurement specialists provided estimates of manufacturer and supplier prices, using available historical data and additional information furnished by specific suppliers. Quality Assurance labor was estimated in accordance with established ratios to direct manufacturing, test and engineering labor hours. All labor was spread in accordance with the program master schedule.

Procurement lead times and subcontractor participation milestones were identified. Labor and material estimates, with schedules, were then provided to cost analysts for determination of costs by application of actual rates and burdens.

Labor burdens were calculated, using current 1970 rates. Labor burdens include overhead, general and administrative expense (G&A) and allocated prime costs (APC). The elements included in these burdens are: (1) Overhead: includes those expenses related to indirect labor cost for plant activities, fringe benefits, shop supplies, facilities including depreciation, utilities, amortization and other labor related costs as established by contractors' accounting policies; (2) G&A: expenses include individual labor costs for company general management, marketing expense, corporate allocations, independent research and development and bid and proposal expenses; (3) APC: includes those costs common to many aerospace programs such as repair and maintenance of manufacturing and test equipment, common engineering technical services, common manufacturing services, common quality assurance services and common computer. Purchased material and services are also burdened at established rates. These burdens include expenses such as procurement cost, receiving inspection and common minor material. Direct labor costs used are the company average at the end of 1970 for engineering, manufacturing, test and remote operations. Direct cost for program management is calculated as a percentage of total direct costs and includes such direct expenses as travel, reproduction, direct per diem and direct supervision. Direct and burden charges were combined to provide average costs per direct manhour, which were used for computing total labor cost. Labor costs were summed with burdened purchased material and services costs to provide total costs by subsystem and by non-recurring, recurring unit and operations costs. Fee was not included to provide consistency with the baseline and target cost data; however, for planning purposes, users of these data could apply fees in accordance with current NASA policies to determine total program resource requirements.

6.3 DEVELOPMENT PLANS

This section summarizes the program plans developed during the study for each of the low-cost spacecraft. These plans, based upon the low-cost designs and the groundrules specified in Section 6.1, were used to develop detailed program costs.

6.3.1 Low-Cost OAO-B Development Plan

The OAO-B program was planned for a four and one-half year span from Phase B start to launch. One year of mission operations following launch is added. Preliminary plans for program management, engineering, test manufacturing and operations were prepared. Sizable reductions in payload costs are offered by use of the new launch vehicles and by application of current technology and practice. Specific cost savings approaches employed are summarized, and detailed cost breakouts to work element are provided in Section 6.4.1.

It is considered that the results are reasonably conservative and that the costs could be further reduced. Some of the conservative approaches applied included:

- A typical NASA phased-program planning approach has been used, with span time of 4.5 years. Based upon LMSC specific experience, this span time can be reduced by over a year with attendant savings in cost. This is particularly true of a Shuttle oriented payload wherein "risks" can be more readily accepted than with conventional expendable-launch payloads.

- Software development has been estimated conservatively.

- Allowances have been made in Manufacturing hours for reasonable rework and scrappage. This is based on actual experience in hardware production, principally in electronics hardware.

- A sizable allowance has been made for Quality Assurance and Inspection. Typical DOD/LMSC rates have been used.

- Hours for test planning and testing have been conservatively estimated.

- Sustaining engineering has been continued between Phases C and D (6 months) per the phased project planning approach.

- Costs have been included in the Shuttle-launched OAO program for two sets of automatic checkout equipment at $2.1 million. This could have been designated as a GFE item at no cost.

- Although LMSC plans include simplified documentation for the low-cost payloads, the actual estimate of the low-cost OAO Program includes full NASA-type documentation and reporting. This was done to eliminate any controversy regarding the feasibility of changing NASA basic procurement policies at this time. However, this area still presents a good target wherein more cost reduction can be obtained if desired.

- A full-qualification test in the form of a shuttle-sortie flight is included.

- The test facilities for the expendable-launched OAO were priced as rental costs for existing facilities. In this way, no new facilities were proposed for the expendable-launched low-cost OAO where in actuality, new facilities may be required, particularly for the future LST, which will not fit into the elaborate existing ground test facilities for the OAO. This biases the cost estimates somewhat in favor of the expendable-launched OAO; here again, changes in assumptions will show the Shuttle-launched OAO more favorably.

- It has been assumed that the Qualification Test Vehicle (QTV) will be produced as a backup flight article for the low-cost OAO. This hardware is actually non-existent in the baseline OAO program. LMSC felt that this approach was a needed insurance for a high-cost development program such as the OAO. However, this is a specific conservatism which is included and represents about $20 million in RDT&E.

- The plan includes 22 man years for handling of off-normal situations which are not pre-planned but which do occur on most programs.

6.3.1.1 Low-Cost OAO-B Program Schedule. The low-cost OAO-B Master Schedule (Fig. 6-2) depicts the major task and subtask phasing, as well as task interrelationships over the life of the program. This schedule reflects the sequence of activities necessary to produce the low cost OAO payload.

The Master Schedule is divided into six major functional areas: Management, Engineering, Manufacturing, Testing, Quality Assurance and Product Support. The interrelationship between these functional areas and the three phases is shown with horizontal bands for the functional areas and vertical divisions for the three phases of procurement.

Management. During the course of the program, all milestones are assumed to be closely controlled to assure adherence to the Program Plan and maintenance of costs. The preparation of the Phase C and D plans plus the Phase D proposal will be major data submissions. Per phased project planning guidelines, it was assumed that contractor funding and level-of-effort will be maintained during the period between Phases C and D.

Engineering. The four design reviews shown in Systems Engineering are key milestones within the program. The concept review will provide direction for the completion of the Phase B Definition Study. The Preliminary Design Review will review the results of Phase B. The Critical Design Review at the end of Phase C will review the final design definition package generated during the design phase. The Acceptance Design Review in Phase D will review the test data and documentation generated during component and vehicle test programs. Based on the results of this review it will be determined if the payload and documentation are acceptable for delivery.

Manufacturing. Manufacturing will provide soft mockup and limited breadboard fabrication support during Phase B. During Phase C, hard mockups plus fabrication and assembly of subsystem components and modules for use in R&D testing

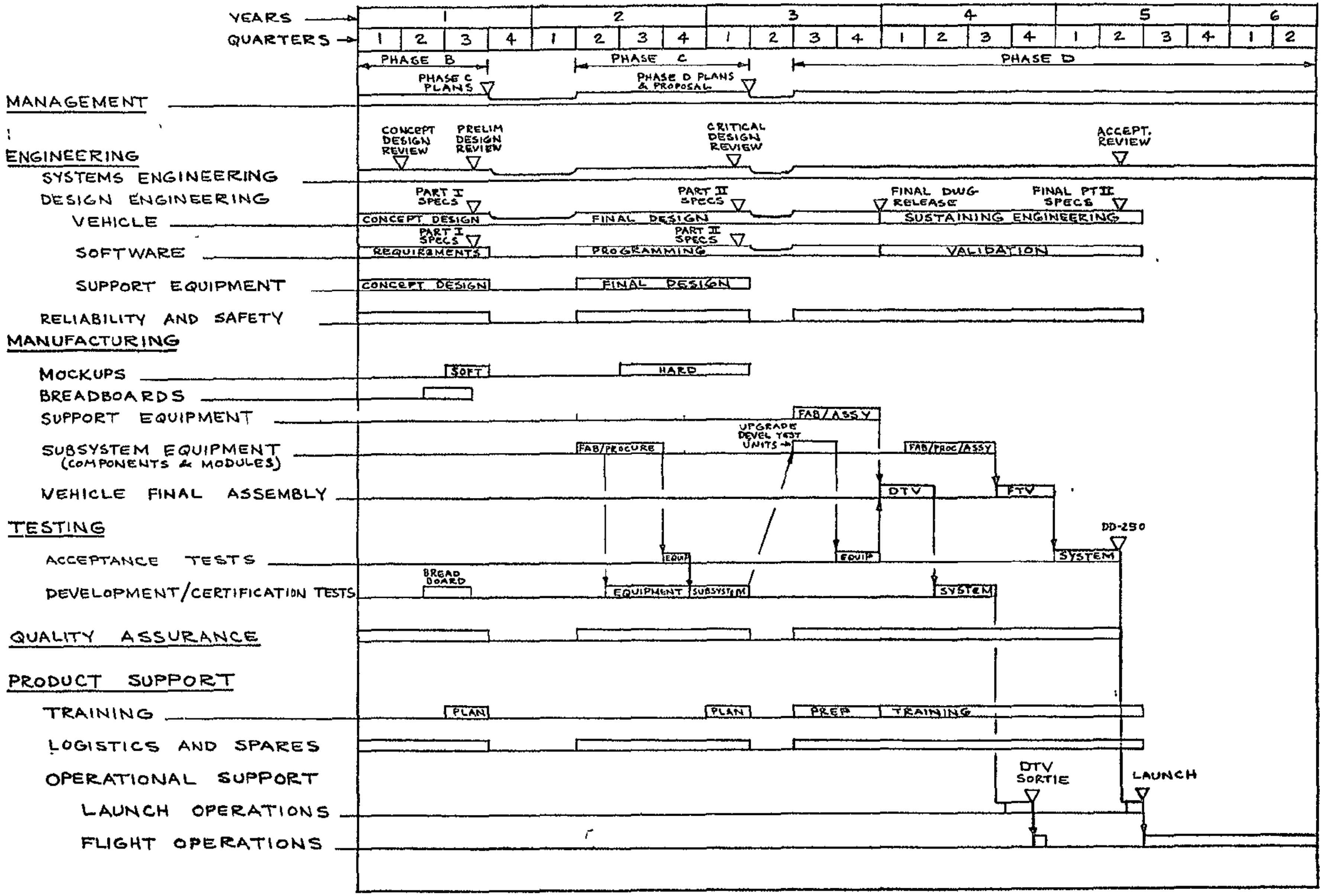

Fig. 6-2 Low-Cost OAO Master Schedule

will be provided. With the approval of the Phase C final design package and the award of a Phase D contract, full production of all tooling, support equipment, components and modules for the Qualification Test Vehicle and the Flight Test Vehicle will commence.

Testing. Development, certification and acceptance testing will occur during the span of the program. The plan provides for use of the hardware used for the Phase C development testing to be used in the assembly of the QTV in Phase D.

Quality Assurance. As with any R&D type unmanned space program, Quality Assurance will be a major consideration throughout the definition, design and development phases of the program.

The approach to the assurance of product quality will be to apply as references and guides portions of NASA NHB 5300.4 (1A & B). In general; any selected and approved variation from these requirements that will not compromise the quality or reliability of the end product, but may significantly decrease and limit the overall level of product verification, and documentation required will be used. Variation will only be permitted if it can be shown to have a significant cost impact with no performance compromise.

Product Support. Space and ground crew training, logistics and operational support will be provided during appropriate phases of the program as shown. Operational support will continue beginning with prelaunch operations, through launch and one-year of operations.

6.3.1.2 Low-Cost OAO-B Engineering Plan. The Engineering Plan was formulated based upon the planned $4\frac{1}{2}$ year program from initiation of Phase B through launch plus one year operational life. The principal engineering functions are Systems Engineering, Design Engineering and Engineering Services. Using NASA Phased Project Planning as a guide to types of work and key milestones, estimates of level of effort and required disciplines were made for input to costing. The general plan, based upon the guidelines in Section 6.1.1, shows

completion of the definition phase, Phase B, in nine months. The outputs of Phase B will be preliminary design data, preliminary systems specifications, preliminary manufacturing and test plans, a preliminary operations and operations support plan, system and subsystem design margins, reliability and quality assurance plans and estimates, the Configuration Management Plan, management plans, schedules and cost estimates, etc., as described in NHB 7121.2, Phased Project Planning Guidelines. Phase B is conducted primarily by the Systems Engineering Organization with support from design, analysis and supporting groups. Phase C, Design, has the objective of overall detailed definition of the project. Outputs are reports, plans and a firm proposal for the Phase D, Development Operations. Principal engineering efforts during Phase C are centered in design engineering under the technical direction of systems engineering and supported by engineering services. Engineering Phase C will result in completed designs and specifications ready for manufacturing and procurement. During Phase D, Development/Operations, engineering will primarily be in support of the manufacturing, test and operations organizations. Some design wrap-up and revision based upon ground and flight test can be anticipated and have been allowed in the estimates. Engineering will support test and operations in evaluation of tests and in conduct of the mission under sustaining engineering. The costs associated with this latter support is included in the recurring unit and operations costs while all of Phases B and C and early Phase D efforts are included in non-recurring RDT&E costs.

6.3.1.3 Low-Cost OAO-B Manufacturing Plan. The Manufacturing Plan was developed based upon the explicit ground rules that there would be one deliverable flight payload and one non-end item qualification test vehicle (the latter to become a backup flight article). Approaches to manufacturing one-only R&D-type payloads impose different requirements than does multiple-unit production-type manufacturing. Additional cost-saving design features that influence manufacturing include relaxed tolerances and material specifications and use of simple straight-forward manufacturing techniques. Interchangeability of modules is required for in-orbit maintenance and refurbishment and necessitates a degree of repeatability in fabrication and assembly of module structures.

Because repetitive production is not required, effective use can be made of a dedicated manufacturing control and planning capability to establish a close, quick reaction operation and to retain experience gained during development for use on the deliverable spacecraft. To this end, a localized and program-oriented operation is planned to provide planning and control, and payload final assembly for low-cost OAO manufacturing. Manufacturing support operations and procedures are selectively applied to optimize cost savings while maintaining effective controls such as the general use of soft tooling except for controlling replacement module interfaces.

6.3.1.4 Low-Cost OAO-B Test Plan. The advent of an operational Space Shuttle permits radical departures from previous aerospace experience associated with expendable launch vehicle programs. The full extent of the Shuttle impact on payload programs can only be predicted at this time, but design concepts will be greatly affected by the reduction of design constraints that led to complicated and sophisticated equipment. The increase in design conservation (higher safety factors) should lead to a reduction in testing because more use can be made of past experience and analyses to achieve confidence in equipment performance without increasing program risks. Assuming equivalent program objectives, reduction of test costs is beneficial to the overall space program.

A low-cost approach to testing must be associated with a reduction in test operations. Reduction implies the elimination of unnecessary (or redundant) tests, or reduction of simulation fidelity for given tests. A reduction of testing in previous aerospace programs to achieve low costs would normally be associated with higher risks in achieving goals of performance, schedule, and costs; which may eliminate any potential cost savings. However, for Shuttle supported launches, these potentially higher risks are alleviated by increasing design conservatism and by the ability to perform on-orbit maintenance and refurbishment. Judicious balancing of these elements (design, test, and on-orbit maintenance) will minimize risk while permitting reduced testing.

6.3.1.5 Low-Cost OAO-B Operations Plan. The use of the Shuttle to deliver the Low-Cost OAO to the desired orbital altitude and inclination, and to recover it if desired, is assumed. Shuttle operations are independent of the payload program except for the initial prelaunch compatibility tests and the cooperative pre-separation payload test phase. The cost of the Shuttle operations are included in the allocated Shuttle user cost. Pre-launch preparation spans are extrapolated from the KSC LC 39 Space Shuttle Test Plan, TR-1078, 4 Nov 1970. Sixteen hours have been added to the KSC schedule to permit combined systems tests, primarily to assure EMI/RFI compatibility, acceptability and data bus interface, and to allow for loading attitude control gas, cleaning and close-out of the cargo compartment. On the launch pad provisions are made for a two-hour period just prior to the Shuttle countdown preparation, during which the payload may be checked out through the Shuttle data bus interface; this corresponds to the OAO "aliveness" test. The payload launch crew, except for a small launch base cadre, is made up of project, subsystem and test personnel who have participated in the assembly and system testing and have followed the "launch-ready" payload to the launch base.

A dedicated control center, staffed by project personnel, with GFE support from the Office of T&DA for NASCOM/STADAN is planned. The same level of NASA T&DA payload support is estimated for low-cost OAO as for current OAO operations; through the lessened dependence upon orbit-by-orbit detail commands (because of the on-board computer) and the ability to store two days' worth of commands suggests the possibility of using fewer stations for STADAN support if other than OAO requirements for a given station were to disappear. Computer support for operations is assumed as GFE because the capability now exists at GSFC and the 5 OAO STADAN stations. One year of Operation Support is included.

6.3.1.6 Application to Low-Cost Expendable Launch Systems.

Design. Initially, the design of the low-cost OAO-B was predicated upon use of the Space Shuttle. Subsequently, design differences to be compatible with the expendable launch vehicles were identified. Principal changes in the design of the Shuttle-launched configuration to the new low-cost expendable (LCE) and

alternate current expendable (ACE) configurations of the low-cost OAO comprised addition of redundant units and backup components to compensate for the inability to revisit, repair and maintain in orbit the expendable-launched OAO payload.

Based upon Shuttle and launch vehicle data provided by the Aerospace Corporation, it was determined that sufficient performance was available, with both the specified LCE, the Seven Segment SRM/Titan Large Diameter Core (TIII-L2) and the planned ACE, the Atlas Centaur, to place over 9,000 lbs (4,082 kg) to OAO's orbit, 400 nm (740 km) at 35° inclination. The Shuttle-launched low-cost OAO-B weighs 7,809 lb (3,542 kg). Changes to this payload to adapt to the LCE or ACE launch vehicles totals 500 lbs (227 kg)(including 143 kg for a structural adapter). This raises the expendable OAO-B's overall weight to 8,309 lb (3,769 kg) which is well within the performance capability of both alternate expendables. As the launch environment is also essentially the same for LCE & ACE, the reconfigurations and redesigns for LCE and ACE are virtually identical. The primary difference between ACE & LCE designs is the requirement for different payload to launch vehicle adapters which in the LCE case mates to a 15 ft (4.57 m) diameter launch vehicle adapter and in the ACE case to a 10 ft (3.05 m) adapter.

Manufacturing. Increased costs reflected in the manufacturing of the expendable OAO are caused by the addition of specific components and units for increased reliability (as contrasted to the "higher-risk" approach taken with Shuttle-launched OAO). Other cost increases were caused by increased levels of manufacturing planning, scheduling and module assembly resulting from increased module complexity. No major cost drivers in manufacturing and assembly of the expendable OAO can be clearly identified because the overall expendable design concept is essentially the same as the Shuttle OAO. It is still planned to develop a QTV for qualification of the total system and to serve as a backup to the FTV.

Testing. Increases in test costs are a result of the increases in spacecraft design complexity plus changes in test philosophy. The spacecraft design for

expendable booster launches includes additional components for redundancy and/or backup, some of which will require new development. Loss of Shuttle capability for predeployment checkout and maintenance causes an increase in pre-acceptance testing to substantiate equipment quality requirements. This is accomplished by increases in the degree of sophistication and extent of qualification testing. Qualification requirements increase costs for test operations and for test hardware above the Shuttle-oriented program. In addition, the use of sequential ground simulation in lieu of Shuttle sortie flights extends the development schedule during a period of high man loading, thereby increasing program costs.

Complete QTV certification is planned which includes system level thermal vacuum, shake, acoustic and stability testing. Elimination of actual flight test of QTV impacts non-recurring test costs; however, operational training and validation of ground system afforded by QTV flight test are no longer available. For the FTV acceptance testing, full ground simulation testing will be required with concomitant increased facility costs.

Operations. Principal changes in the operational costs center around the preparation for launch of the expendable OAO. Although these cost changes are relatively minor, allowances have been included for increased system validation at the launch base and for integration of payload with the launch vehicle. Increased logistic costs reflect the need to transport the adapter to the launch base. Use of the QTV for launch base certification and training is included. Costs and time for mating the adapter with the launch vehicle, and for joint flight acceptance testing (J-Fact) have been included. Due to the fact that both systems are designed for unattended operation, there are no measurable changes in the costs of mission operations. The mode but not the cost of orbital checkout and system verification prior to initiating operations will differ in that no shuttle crew is available to assist in these activities.

The comparative cost advantages of the Shuttle retrieval of a malfunctioning payload have not been included.

6.3.2 Low-Cost SEO Development Plan

The SEO program for the Shuttle/Tug was planned for a three and one-half year span from start of Phase B to the launch of the first operational SEO. The time from hardware go-ahead (start of Phase C/D) to first launch is 30 months, closely approximating the Lunar Orbiter schedule. The program objective is to provide continuous earth observation with a network of four satellites for a period of over ten years. The nominal spacecraft lifetime is two years, which necessitates replacement or refurbishment on a two-year cycle. With Shuttle/Tug, replacement is accomplished by delivering a replacement spacecraft to SYNEQ orbit station, retrieving the malfunctioning system and returning to low-earth orbit for rendezvous and recovery with the Shuttle. Included in the plans (and costs) are the full development of the system, launch and initial placement of four SEOs in orbit and two years of operation. Five operational spacecraft are planned with the fifth serving as a backup and initial replacement system. As with OAO, the spacecraft features replaceable modular subsystems for low-cost assembly and refurbishment. A development test vehicle (DTV) is planned for initial R&D development testing; the DTV will be upgraded to a Qualification Test Vehicle (QTV) for use in system qualification testing, training, operational readiness demonstrations and launch interface testing. A complete operational network using 4 existing tracking stations and a mission control center (MCC) is provided and operations for the first two years following initial launch are costed.

A plan for development of the expendable SEO using the Titan IIID/Centaur was also prepared. In the expendable launch system, refurbishment is not possible; replacement means launching a new spacecraft, and additional lifetime and performance confidence is required. A four year program is necessary because of additional qualification and reliability testing. Although the designs of the expendable and reusable SEOs are virtually identical, the additional testing raises RDT&E costs.

6.3.2.1 Low-Cost SEO Program Schedule. The SEO Master Schedule (Fig. 6-3) depicts the major task and subtask phasing, as well as task interrelationships

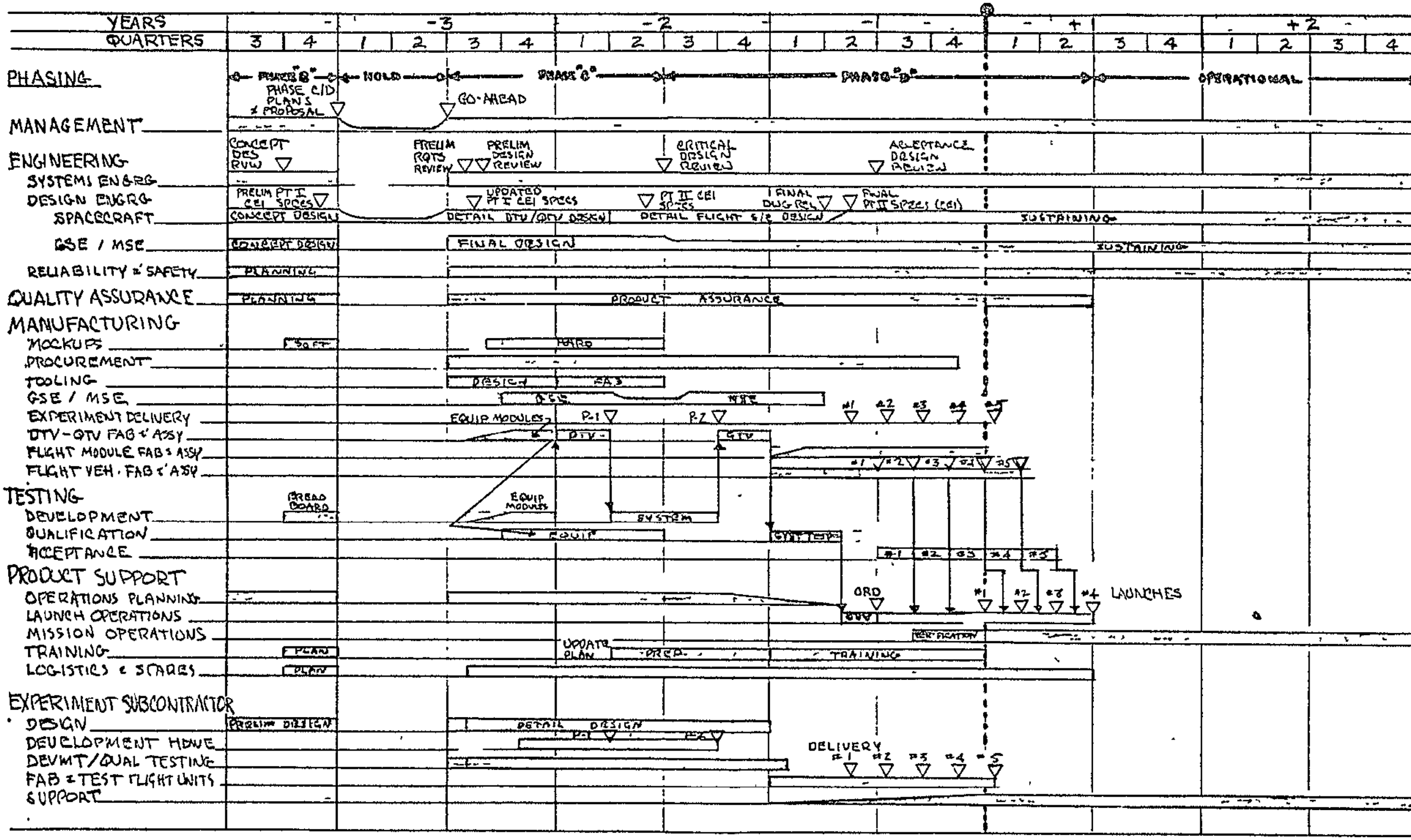

Fig. 6-3 Low-Cost SEO Master Schedule

over the life of the program. This schedule reflects the sequence of activities necessary to produce the low cost SEO payload.

The Master Schedule is divided into seven major functional areas: Management, Engineering, Quality Assurance, Manufacturing, Testing, Product Support and Experiment Subcontractor. The interrelationship between these functional areas and the three phases is shown with horizontal bands for the functional areas and vertical divisions for the three phases of procurement.

Management. During the course of the program, all milestones are assumed to be closely controlled to assure adherence to the Program Plan and maintenance of costs. The preparation of the C/D plans and proposal will be major data submissions.

Engineering. The four design reviews shown in Systems Engineering are key milestones within the program. The concept review will provide direction for the completion of the Phase B Definition Study. The Preliminary Requirements Review will review the results of Phase B and the Preliminary Design Review will review the solidified design approach. The Critical Design Review at the end of Phase C will review the final design definition package generated during the design phase. The Acceptance Design Review in Phase D will review the test data and documentation generated during component and vehicle test programs. Based on the results of this review it will be determined if the payload and documentation are acceptable for delivery.

Manufacturing. Manufacturing will provide soft mockup and limited breadboard fabrication support during Phase B. During Phase C, hard mockups plus fabrication and assembly of subsystem components and modules for use in R&D testing will be provided. With the approval of the design package, full production of all tooling, support equipment, components and modules for the Qualification Test Vehicle and the Flight Test Vehicle will commence.

Testing. Development, certification and acceptance testing will occur during the span of the program. The plan provides for use of the hardware used for

the Phase C development testing to be used in the assembly of the QTV in Phase D.

Quality Assurance. As with any R&D type unmanned space program, Quality Assurance will be a major consideration throughout the definition, design and development phases of the program.

The approach to the assurance of product quality will be to apply as references and guides portions of NASA NHB 5300.4 (1A & B). In general, any selected and approved variation from these requirements that will not compromise the quality or reliability of the end product, but may significantly decrease and limit the overall level of product verification, and documentation required will be used. Variation will only be permitted if it can be shown to have a significant cost impact with no performance compromise.

Product Support. Space and ground crew training, logistics and operational support are provided during appropriate phases of the program as shown.

Experiment Subcontractor. Required design, development, fabrication, test and support spans and milestones for the experiment subcontractor are shown.

6.3.2.2 Low-Cost SEO Engineering Plan. The Engineering Plan was formulated based upon the planned $3\frac{1}{2}$ year program from initiation of Phase B through launch plus two year operational life. As with OAO-B, the principal engineering functions are Systems Engineering, Design Engineering, and Engineering Services. Using NASA Phased Project Planning as a guide to types of work and key milestones, estimates of level of effort and required disciplines were made for input to costing. The general plan, based upon the guidelines in Sections 6.1.2, and upon the Lunar Orbiter Program, shows completion of the definition phase, Phase B, in six months. There was no Phase B during Lunar Orbiter due to the input need for early flight readiness. The outputs of Phase B will be preliminary design data, preliminary systems specifications, preliminary manufacturing and test plans, a preliminary operations and operations support plan, system and subsystem design margins, reliability and quality assurance plans

and estimates, the Configuration Management Plan, management plans, schedules and cost estimates. Phase C, Design, has been combined with Phase D, Development Operations in order to maintain the similarity to Lunar Orbiter. It is assumed the Prime Contractor selection will occur between Phase B and Phase C/D.

Qualification and acceptance testing can be anticipated and have been allowed in the estimates. Engineering will support test and operations in evaluation of tests and in conduct of the mission under sustaining engineering. The costs associated with this latter support is included in the recurring unit and operations costs while all of Phases B and C and early Phase D efforts are non-recurring RDT&E.

6.3.2.3 Low-Cost SEO Manufacturing Plan. Essentially the same manufacturing approach used for the low-cost OAO-B will be applied to the low-cost SEO. However, the SEO is fundamentally a smaller, simpler payload; for example: (1) The precise alignment of the OAO telescope is not required; (2) the general tolerances for boresighting of sensors are relaxed; (3) the SEO structure is less complex, contains fewer detail parts, and weighs less than OAO. The use of commonly available materials and standard shapes and sizes also reduces manufacturing requirements and costs. The SEO production quantity and rates justify reasonably sophisticated tooling to control the alignment, assembly and attachment of the interchangeable subsystem modules, which require easy removal and installation (for orbit repair operations). Considerable savings are gained by using class "A" tooling (cost amortized over several units of hardware) and in turn lowering the man-hours spent in fabrication and assembly. There are no complex or sophisticated structural manufacturing problems that would affect function or schedule.

Significant cost reductions are obtained in manufacturing the electronic components (black boxes) by minimizing material costs and labor as a result of standardizing mounting compartments and mounting racks, connectors, printed circuit boards, circuitry elements and modules. The packaging density of components will be reduced considerably which permits ease of assembly, inspection, rework, etc.

6.3.2.4 Low-Cost SEO Test Plan. Prior approaches to low-cost test programs will be used for SEO as modified for its production and schedule requirements. Because nearly 10 sets of components are required, sophisticated, dedicated support equipments can be justified. Both of these factors increase engineering and equipment costs to permit reduction in recurring test costs. In addition, operational consistency provided by control and standardization enhances the confidence in the quality of subsequent production based upon one-time qualification.

Although the SEO approach to development testing is consistent with previous low-cost programs, the amount of testing is reduced because there are many off-the-shelf equipments available and new developments are well within current technology and production capabilities. For this reason, even initial development efforts are directed toward final equipment packaging. Most of the equipment for the DTV can be procured to final configuration, and component qualification can proceed in parallel with initial system integration tests. Component qualification will develop high confidence in operational deployment to assure that either the flight vehicle or on-board spares survive the launch environments with on-site maintenance as required. Although formal qualification is required for SEO, maximum use will be made of previous tests and analyses to substantiate quality without testing. Where tests are required, one unit will suffice.

The most significant change on the SEO is the subcontracted photographic experiment. The extent of development requires that this effort precede the basic spacecraft development. Interface requirements between the photographic module and the spacecraft are simple and minimum to permit parallel development. Even so, the subcontractor is required to provide units early in the spacecraft development cycle to assure delivery capability and systems compatibility early in the program.

Acceptance testing is consistent with previous low-cost concepts which verify quality at the component, module and system levels. Because of the delivery schedules, two system test complexes are required to process spacecraft concurrently.

During development environmental stressing is performed at the module level to take advantage of the common modular interfaces and to perform concurrent testing of components. Where new equipments are being developed, some environmental stressing is performed at the component level to reduce downstream risk. For qualification and acceptance, environmental stressing is performed at the component and spacecraft levels. Components are exposed to thermal stresses and vibration environments. The spacecraft is exposed to thermo-vacuum and a coustic environments.

Sophisticated, dedicated GSE and STE are justified for SEO based on production quantities and rates. STE, which is used in-house by the contractors, will be controlled to less formal procedures than GSE which is delivered to NASA for operations. Automatic test stations will be used for component and module testing. System testing will be accomplished using an automatic, standardized test station developed independently and provided to the program for a fixed users fee. The increased RDT&E cost for GSE, STE and handling equipment is offset by reductions in recurring cost achieved by simplicity and speed of operations. Scheduling of end-to-end testing by the same test personnel for sequential production permits effective learning carry-over to further reduce cost and enhance quality. Functional test equipment is provided to simulate interfaces between the spacecraft/shuttle/tug and spacecraft system to photographic experiment.

6.3.2.5 Low-Cost SEO Operations Plan. Space Shuttle and tug operations are independent of the SEO program operations except during the brief pre-launch joint flight acceptance composite tests (J-FACTS) and the coordination during on-orbit tests with the Shuttle orbiter prior to separation. The use of a Standard Payload Checkout Set installed in the Orbiter cargo bay to check out both the tug and the SEO requires that both be designed with this capability as a firm requirement; the versatility and cost advantages are such that this requirement should be considered valid.

Five SEO spacecraft are provided for a 4-station system. Following the Lunar Orbiter precedent, a backup launch-ready spacecraft has been planned for each

initial-placement launch. Should schedules prove too tight, this requirement could be waived, thus gaining two months for the manufacturing, integration and test activities. Because of the repetitive launches and the need for maintaining readiness for replacements, the launch operations activity for SEO are planned to continue for the program duration.

A dedicated control room equipped to handle five SEO satellites is part of the Mission Control Center, assumed to be located at the GSFC, Greenbelt, Maryland. Four existing NASA ground stations will each provide an S-band two-way link to its assigned SEO; a fifth, the KSC station, is used for pre-launch and launch checkout and as a backup to the Rosman, North Carolina, station for LEO checkout. Station operations and communications from the remote sites to MCC is provided GFE by the NASA Office of Tracking and Data Acquisition. Some SEO-peculiar equipment and a small cadre of SEO specialists will be located at each station, thus providing the capability of continuing SEO operations during temporary interruptions in the communications links to the MCC.

6.3.2.6 Application to Low-Cost Expendable Launch Systems.

Design. The low-cost SEO, derived from Lunar Orbiter, was first designed for launch on Space Shuttle/Space Tug. For SEO, the Titan IIID/Centaur was designated by Aerospace as the new Low-Cost Expendable (LCE) launch system. A re-examination of the SEO low-cost design indicated that only minor modifications were required. The principal design differences include:

- Addition of Launch Vehicle/Payload Adapter
- Addition of Second Redundant Transponder
- Addition of a Redundant TWTA
- Addition of a Separation Ring
- Deletion of the Docking Ring
- Deletion of the Docking Reflectors

Schedule. The program master schedule was increased by six months during the development phase (Phase D) to assure time to accomplish additional requisite reliability and qualification testing. An additional QTV was added for use in this testing. The additional testing is required to provide an adequate confidence level necessary in the 2-year mission in the expendable mode. With Shuttle considerable testing was eliminated because of the system confidence gain afforded by revisit, repair, and refurbishment. The additional test equipment and articles and additional manpower required, coupled with the longer development span, have increased program RDT&E costs measurably.

Manufacturing. Manufacturing the SEO for the Low-Cost expendable launch system will be practically the same as for the Shuttle/Tug. Principal design changes in the adapter and separation ring and the added redundant components offer no significant problems. Fabrication and assembly of the additional QTV will raise RDT&E costs and lengthen the schedule.

Testing. Changes in the approach to testing resulting from using a low-cost, expendable booster instead of Shuttle are primarily caused by the loss of predeployment checkout and in-orbit repair. To counteract the loss of this feature, confidence in achieving successful deployment must be generated through prior ground testing. In both cases, the inherent quality of design and production is the same, but the impact of failure is less with Shuttle launches. Traditionally, the level of confidence has increased with the amount and fidelity of testing, and such changes are implemented in the low-cost, expendable booster program. To this end, a Qualification Test Vehicle (QTV) composed of prototype equipment is required in addition to the Development Test Vehicle (DTV), which is still planned for subsequent upgrading. Since the tests were previously planned, the cost impact is primarily in increased hardware requirements. In addition, the adapter and separation system requires development, and increases in CDPI subsystem equipment increases the complexity of system testing.

Operations. The basic operations plan is only moderately affected by the substitution of a Titan IIID/Centaur for the Space Shuttle because the ground rules called for maintenance of a flight-ready backup spacecraft in both cases. The preflight operational checkout is more detailed and repetitive since a much higher confidence level is required, and the in-flight post-ascent shuttle checkout mode is not possible. Standard Payload Checkout Sets are still applicable, with supplementary stimuli (i.e., horizon sensor targets), possibly indicated. The Titan uses the Integrate-Transfer-Launch (ITL) launch operations mode, with buildup in the ITL and movement of the launch vehicle to Pad 41 on rails. Installation of the SEO can be accomplished either in the VIB or at the pad, but in either case a checkout van is required to house the payload checkout equipment. As with shuttle, flight operations up to completion of injection are the responsibility of the transportation system, with the payload (SEO) in a standby mode. Ascent tracking and readout of SEO telemetry is optional but not considered necessary. Mission operations are the same as for the Shuttle mode following placement in orbit.

6.3.3 Low-Cost SRS Development Plan

The SRS program was planned for a two-year span from go-ahead to first launch. The HIGLO mission is to obtain two or more years of continuous observation of the earth's magnetosphere in a polar, eccentric sun-synchronous orbit. For the Shuttle launch, it was planned that three spacecraft would be carried as piggyback payloads during launch. In orbit, all three would be checked out and two launched. The first would commence operation with an expected lifetime of six months. The second spacecraft would place itself in a higher orbit and would go dormant for six months (or until required). Upon wearout of the first spacecraft, the second would be activated and would continue the program of observations. The third (unused) payload would be returned to earth in the Shuttle for later use.

The cost of SRS does not lend itself to economical recovery and refurbishment from orbit; hence SRS is treated as an expendable payload. In the expendable

launch vehicle (ACE/LCE) mode, two satellites are launched on a single expendable launch vehicle. The dual mission mode is virtually identical to the Shuttle mission-mode except that a third payload is not available for substitution during initial orbit placement in the event that one failed to survive ascent.

As the P-11 was developed as a low-cost piggy-back payload for a series of experimental quick-response programs, many of the conventional NASA program management and testing approaches were not applied. This resulted in sizable program cost savings. As similar program approaches were used in planning the low-cost SRS program, potential users of these data are cautioned that costs for similar programs utilizing conventional NASA and DOD program guidelines would be greater.

Cost saving design approaches developed for low-cost OAO and SEO were generally applied except that use of in-flight replaceable modules was not required. However, modular arrangement of equipment, for reducing manufacturing, qualification and testing costs, was used.

A major programmatic variation is introduced by the Shuttle in that the first production spacecraft is used for qualification and reliability testing and is subsequently reworked to serve as the fourth flight article. This resulted in a savings of approximately $1.5 million in RDT&E costs. For the expendables, a full Qualification Test Vehicle is provided for qualification and reliability testing. The inclusion of this additional qualification hardware to the RDT&E phase would increase RDT&E costs for the Shuttle program to $8.24 million and effectively reduce overall program savings. However, it is considered that the approach used is valid in that it is considered that no spare is required in the Shuttle mode because retrieval from orbit is possible in the event that one of the three payloads carried fails during ascent. In the expendable and baseline modes, although the Qualification Test Vehicle is not reworked to flight ready condition, it is available as a spare flight article for use in the event of failure of one of the operational vehicles. This would require rework and

acceptance testing, however, and would incur additional costs that are not currently included in the expendable configuration modes.

Percentage savings are not as large on the SRS payloads as on the other programs studied. This is attributable to the fact that SRS baseline costs are very low without payload effects. However, savings were realized by applying the techniques developed during the study.

6.3.3.1 Low-Cost SRS Program Schedule. The SRS Program Master Schedule, Fig. 6-4, depicts the major task and subtask phasing planned for the Shuttle-launched Low-Cost Small Research Satellite. It also shows task interrelationships and hardware flow for the span of the program through the launch of the second pair of spacecraft. The plan is divided into the five major work elements, Engineering, Quality Assurance, Manufacturing, Testing and Product Support. NASA-phased project planning guidelines were not directly used in developing this low-cost payload. Instead it was assumed that program go-ahead would follow an industry competition. There are three overlapping phases planned: Design and Development commences with go-ahead and ends with acceptance of the first flight vehicle. Production commences with fabrication and subassembly of flight payload number 1 at ten months after go-ahead and ends with the completion of refurbishment of FTV-1 (previously used for qualification and reliability testing) at one month prior to first launch. Operations support commences with acceptance of the first payload to be launched (FTV-2) and continues through the balance of the program.

6.3.3.2 Low-Cost SRS Engineering Plan. Principal changes in the engineering approach are summarized in the ground rules in Section 6.1.3. Critical milestones for engineering are the submission and approval of the preliminary specifications at two months after go-ahead. Engineering and design proceeds towards the submission of the detailed specifications at the start of the sixth month. At this time the initial critical design review will be conducted. Configuration freeze occurs nine months after go-ahead coincident with the start of fabrication of FTV-1. A hard mockup will be developed early in the program for use as a design aid. The final critical design review occurs at the

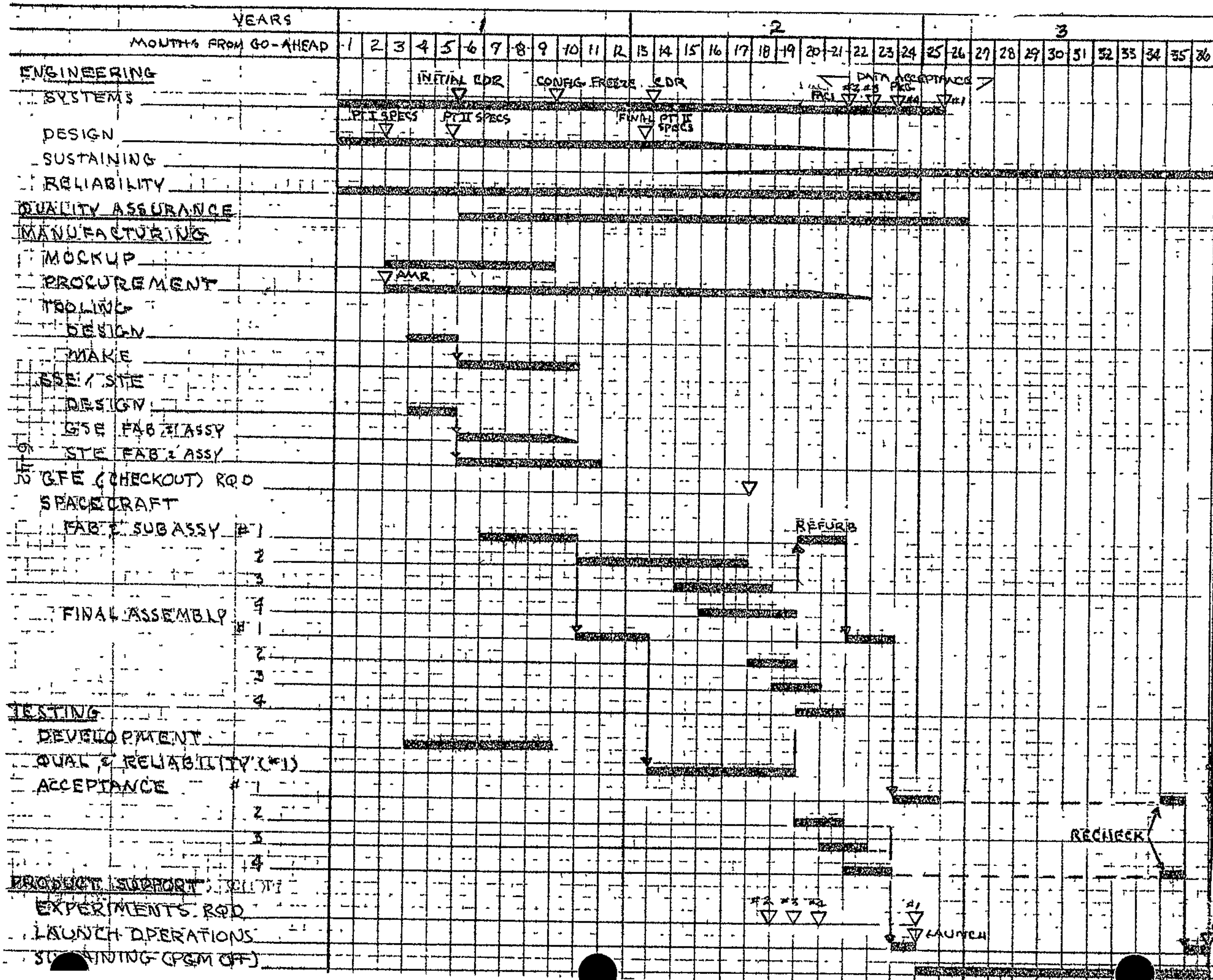

Fig. 6-4 Low-Cost SRS Schedule

thirteenth month and coincides with the submission of the final specifications and with the initiation of qualification and reliability testing on FTV-1. Engineering and design support to manufacturing and test continues through the production phase. A small sustaining engineering program office will be maintained following completion of the design and development phase to support the government in operating the payloads during launch and mission operations.

6.3.3.3 Low Cost SRS Manufacturing Plan. Manufacturing starts with the fabrication of the mockup and continues through the refurbishment of FTV-1 following reliability testing. Tooling and support equipment fabrication starts following the initial critical design review. Support equipment must be completed in time to support final payload assembly and acceptance testing. Final assembly spans are shortened because of the modular design of the low-cost SRS. Fabrication and assembly time for electronic components is shortened due to the low-density packaging. Procurement starts early in the program for mockup materials. Long-lead hardware procurement starts at initial CDR.

6.3.3.4 Low-Cost SRS Test Plan. Testing commences with development testing in support of the payload subsystem designs. Development testing will continue through to configuration freeze at nine months. Qualification and reliability (lifetime) testing continues for six months and ends coincident with the start of acceptance testing on FTV-2. Acceptance testing will include system verification, interface verification with the Shuttle and its on-board checkout system, anechoic testing, magnetic cleanliness testing, acoustic testing at the Shuttle ascent environment levels, experiment interface testing and full thermal-vacuum testing. Integration of the experiment package will occur during final assembly and the experiment package will undergo complete testing with the spacecraft. Delivery of experiment packages must coincide with this schedule as is shown in the Master Schedule.

Following acceptance testing, the final acceptance design review is scheduled and the acceptance data package is delivered. Coincident with the final FTV-2 acceptance review, the First Article Configuration Inspection (FACI) will be conducted. FACI has the objective of comparing the deliverable hardware to

the design drawings and specifications. The testing phase concludes with the acceptance testing of the reworked FTV-1.

6.3.3.5 Low-Cost SRS Operations Plan. As stipulated in the ground rules and consistent with the practice on P-11, operations are conducted by the government with support from the program office. No direct contractor participation, except as required to support the government, is planned for launch operations. As experiment data can be easily recorded for delivery to the users, and as the payload is relatively easy to control, mission operations support by the contractor is limited to an advisory capacity. Procedures for launch operations and checkout and for mission operations will be developed during the development phase and delivered to the government at the first acceptance design review.

6.3.3.6 Application to Low-Cost Expendable Launch Systems. As the baseline P-11 spacecraft could not supply all of the required support to the HIGLO mission in terms of stabilization method, attitude control, power availability and communications capacity, modifications to the designs were required. These design changes were made with the support of the P-11 program designers and are described in Section 5.4. Changes to the Shuttle launched SRS for launch by ACE/LCE include the provision of a launch vehicle/SRS adapter, addition of an auxiliary flight control electronics unit to backup the primary attitude sensing equipment and to increase confidence in mission success lost by not being able to checkout the payload in orbit. Redundant equipment were added to the CDP&I and the Electrical subsystems to improve overall reliability for the same reasons.

The principal change in program planning was the addition of a separate Qualification Test Vehicle (QTV) to the expendable program. Besides providing additional confidence in the design and operation of the payload, it could serve as a backup FTV with rework. This backup is not necessary with the Shuttle as a malfunctioning payload could be retrieved from orbit prior to deployment and mission commitment. It is not planned that the SRS could be retrieved in the

Shuttle mode, however, retrieval could be possible during placement of subsequent payloads providing there is no interference with primary payload operations.

Production and testing of the QTV would take place at the time indicated in the SRS/Shuttle Schedule (Fig. 6-4) for manufacture, assembly and qualification testing of Shuttle FTV-1. The overall schedule effect would be to extend the production and test phase one month with the completion of acceptance of the fourth FTV at 25 months. For the expendable mode, some schedule slips on FTV 3 & 4 are permissible as only two are required for initial launch instead of three as indicated for Space Shuttle.

6.3.4 Recosted Baseline

In order to provide visibility into the magnitude of the payload effects and to provide additional data to support the credibility to the study results, NASA requested the study to recost the baseline OAO-B and SEO programs to the same level of detail as the low-cost programs were estimated. Specific ground rules for this recosting were approved by NASA, plans and schedules were developed, and program costs were estimated.

This section outlines the ground rules and presents the resulting schedules. Development hardware was summarized in Fig. 6-1. The costs are summarized in Section 6.4 and compared with the low-cost estimates in Section 6.5.

The agreed groundrules are summarized below:

6.3.4.1 OAO-B Recosted Baseline Ground Rules and Schedule. The groundrules for the recosting of OAO-B baseline were:

- Baseline OAO mission and experiment objectives are unchanged.
- Configuration of flight spacecraft (FTV) will be identical with the baseline OAO-B as launched; there will be no changes in design nor application of 1970 state-of-the-art improvements.

- All price estimates will be made using average contractor rates and 1970 dollars.

- NASA phased project planning guidelines (NHB 7121.2) will apply. The program will commence with Phase B.

- A prime aerospace contractor will conduct the program under the direction of a NASA project office.

- Government furnished equipment, facilities and services include: launch vehicles, exit fairings, launch services, control center services, STADAN services, NASCOM, and use of the NASA operational computer.

- The launch vehicle is the SLV-3C/Centaur.

- A single primary FTV will be built, tested, delivered and launched.

- A Qualification Test Vehicle (QTV) will be fabricated and used for development and qualification testing and will serve as a backup FTV following modifications. Development and qualification testing will be done at contractors facility.

- Ground support and mission support equipment equivalent to baseline will be developed.

- All computer programs will be considered new developments. Software will be developed to a degree of equivalence to that produced during the baseline program.

- Planning, reporting and management procedures will be consistent with current NASA practice for unmanned R&D space missions. Quality assurance and reliability programs will be consistent with NHB 5300.4 (A&B).

- The experiment development is part of the overall development and will be costed on the same basis.

- There will be two additional complete sets of flight-type components provided; one for component qualification and one for component reliability testing. Also, a set of development hardware comprising those components that are new developments (not off-the-shelf) will be provided for development test.

- A structural test article will be built.

- A second structural test article will be fabricated for subsequent rework and assembly as the Qualification Test Vehicle (QTV). This test article will initially serve as a Development Test Vehicle (DTV).

- All program-developed GSE and MSE for support of launch and mission operation will be deliverable items and will undergo acceptance testing.

- Spares and logistics costs will be calculated on the same basis as the low-cost OAO-B.

- The FTV acceptance will be accomplished at the Contractor's Facility: The FTV will be shipped to KSC for launch. This is a sizable variation from the OAO-B plan wherein the FTV was shipped to GSFC as GFE for final experiment integration, completion of qualification testing and integrated systems and environmental testing. Costs for these in-house GSFC operations will not be shown, but will be replaced by equivalent costs for contractor in-house operations.

- GSE, MSE, and computer programs will be delivered to the using location for testing, validation and government acceptance.

- The backup FTV (QTV) will be retained at the contractor's facility in flight ready status for a maximum of 3 months after launch for subsequent government direction/disposition.

- Contractor support will continue through the flight operation phase. (One year subsequent to launch).

- NASA project and project support (OT&DA) personnel costs are not included.

- Prime contractor fee will not be included.

- The costs include all elements shown in the low-cost OAO-B costs.

The schedule resulting from the recosted baseline analysis is provided in Fig. 6-5.

6.3.4.2 <u>SEO Recosted Baseline Ground Rules and Schedule</u>. The groundrules for the recosting of the SEO Baseline were:

- Baseline SEO mission and experiment objectives are unchanged.

- Configuration of flight spacecraft (FTV) will be as close as possible to the baseline Lunar Orbiter with changes only as dictated by the SEO mission; there will be no changes in design nor application of 1970 state-of-the-art improvements: spacecraft design emphasizes high reliability, long life and minimum weights.

- All price estimates will be made using average contractor rates and 1970 dollars.

- NASA phased project planning guidelines (NHB 7221.2) will apply. The program will commence with Phase B.

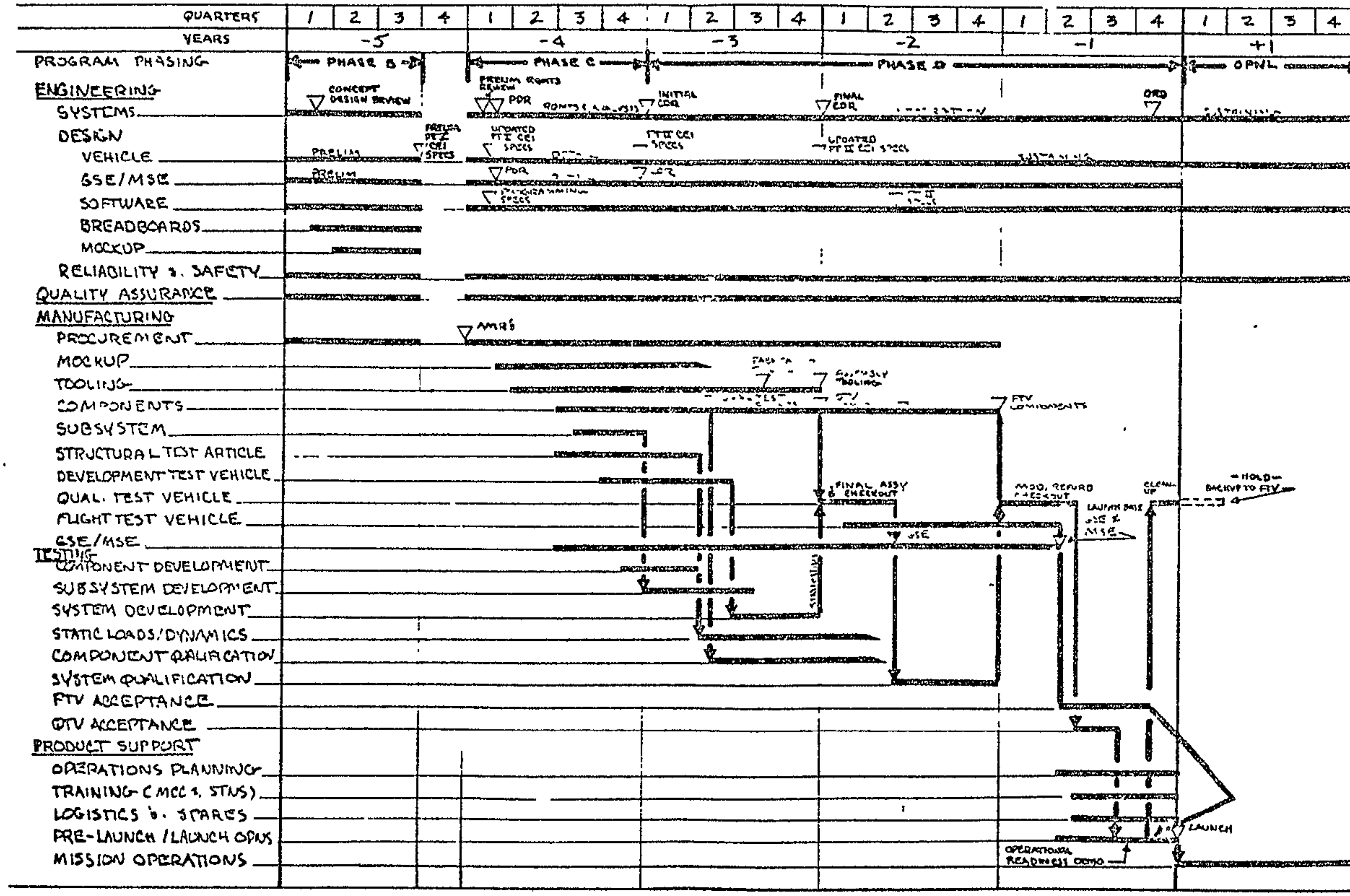

Fig. 6-5 OAO-B Baseline Master Schedule

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- A prime aerospace contractor will conduct the program under the direction of a NASA project office.

- Government furnished equipment, facilities and services include: launch vehicles, exit fairings, launch services, control center services, STADAN services, NASCOM, and use of the NASA operational center.

- The launch vehicle is the SLV-3C/Centaur/Burner II.

- Four primary flight test vehicles (FTV) will be built, tested, delivered, and launched at two month intervals. Also, one additional FTV will be built, tested, and delivered as a backup vehicle.

- Two qualification test vehicles (QTV) will be fabricated and used for qualification testing. Also a hard mockup and 2 structural test articles (STA) will be built.

- 3 sets of ground support equipment will be developed with 2 being utilized for factory testing and the third for support of launch operations (the latter is included in the recurring unit costs).

- All computer programs will be considered new developments. Software will be developed to a degree of equivalence to that produced during the baseline program.

- Planning, reporting and management procedures will be consistent with current NASA practice for unmanned R&D space missions. Quality assurance and reliability programs will be consistent with NHB 5300.4 (A&B).

- The experiment development is part of the overall development program. Similar to Lunar Orbiter, the responsibility is assigned

to a camera subcontractor. In addition to the five flight cameras and the two qualification cameras, three prototype cameras will be built by the camera contractor.

- There will be three additional complete sets of flight-type components provided; one for component qualification, one for component reliability testing, and one for development testing.

- Mission support equipment will be developed identical to those for the low-cost design.

- All program-development GSE and MSE for support of launch and mission operation will be deliverable items and will undergo acceptance testing.

- Spares and logistics costs will be calculated on the same basis as the low-cost SEO.

- The FTV acceptance will be accomplished at the contractor's facility; the FTV will be shipped to KSC for launch.

- Contractor support will continue through the flight operation phase. (Two years subsequent to launch of the first spacecraft.)

- NASA project and project support (OT&DA) personnel costs are not included.

- Prime contractor fee will not be included.

- The costs will include all elements shown in the low-cost SEO costs.

The schedule resulting from the recosted SEO baseline analysis is shown in Fig. 6-6.

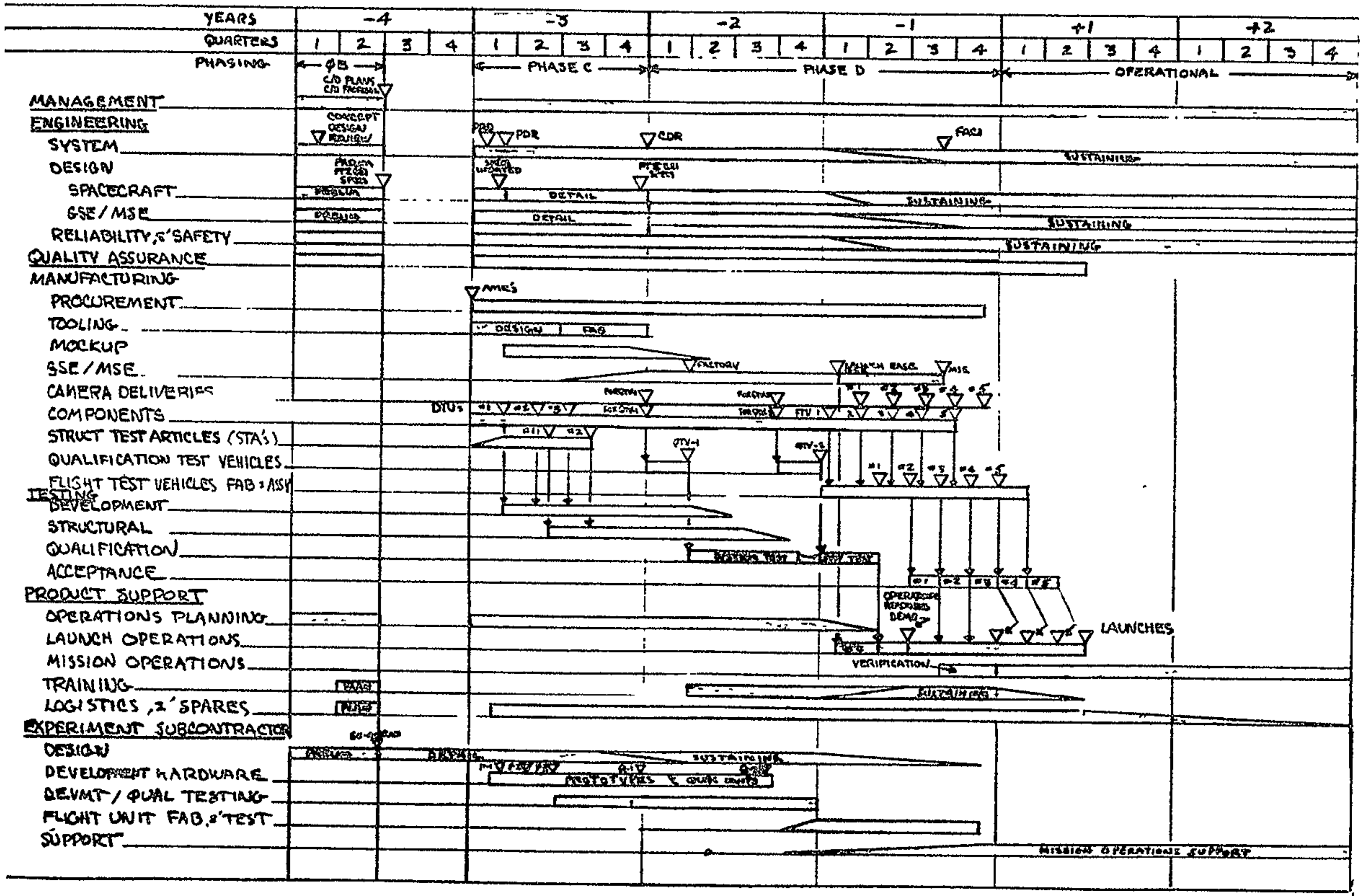


Fig. 6-6 SEO Baseline Master Schedule

6.4 COST SUMMARIES AND COMPARISONS

The cost estimates are presented for the OAO-B, SEO, and SRS payloads, Shuttle or expendable launched. The recosted baseline data for the OAO-B and SEO are included in detail. For the SRS, adjusted baseline costs are shown. SRS was not subjected to the complete recosting as were the previous two spacecraft.

Included in the results are comparisons between the baseline and the low-cost payload cost estimates and time-phased annual funding for these payloads. Since the SRS payload was subjected to less concentrated analysis, funding spreads were not prepared. Furthermore the SRS cost estimates are not presented to the same level of detail as OAO-B and SEO.

6.4.1 OAO-B Cost Estimates

The OAO-B payload costs represent the total development and operation of one flight spacecraft for a one-year program duration.

6.4.1.1 Shuttle-Launched OAO-B. Figures 6-7 and 6-8 show the low-cost OAO-B estimates for the Shuttle-launched case. The RDT&E cost of $84M represents complete payload development including a Shuttle-sortie flight test costing $5M. The recurring costs shown are for a single unit production $15.8M, and one year mission operations of $5.3M.

The cost breakdown by subsystem is in Fig. 6-7 with the functional cost breakdown shown in the Payload Summary, Fig. 6-8. All costs shown are in 1970 dollars and include costs non-allocated to a subsystem. In the development of these data functional cost breakdowns by individual subsystems were produced, but are not included herein.

Figures 6-9 and 6-10 present the time-phased costs for the OAO-B/Shuttle. Both the annual funding and totals by development phase are shown spread over a planned $5\frac{1}{2}$ year span including one year of operations.

SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS

(1970 $ THOUSANDS)

PAYLOAD: OAO-B LAUNCH VEHICLE: SPACE SHUTTLE FLIGHT DURATION: 1 YEAR

SUBSYSTEM	COSTS			
	RDT&E COST	UNIT COST	UNIT OPERATIONS	TOTAL COST
PAYLOAD ASSEMBLY & INTEGRATION	—	—	—	—
EXPERIMENTS	$ 9910	$ 2910	$ 796	$ 13616
STRUCTURES & MECHANISMS	6688	947	150	7785
ELECTRICAL & PYROTECHNICS	11240	2807	720	14767
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL	26996	4669	1189	32854
PROPULSION & ATTITUDE CONTROL	4296	1014	176	5486
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)	20903	2748	2125	25776
ENVIRONMENTAL CONTROL	2899	394	73	3366
NON-ALLOCATED COSTS	1100	325	120	1545
PAYLOAD TOTAL	$ 84032	$ 15814	$ 5349	$ 105195

Fig. 6-7

DETAILED COST ALLOCATION TO EACH SUBSYSTEM
(1970 $ THOUSANDS)

SUBSYSTEM PAYLOAD SUMMARY * PAYLOAD TYPE OAD-B SHUTTLE

	COST CATEGORY	LABOR ENG.	LABOR MFG.	LABOR PA	MATERIEL MATERIAL	MATERIEL EQUIP/PARTS	SUB-TOTALS	TOTALS
RDT&E	PROGRAM MANAGEMENT	$ 4392	$ 408	$ 451	$ 48	$ 1163	$ 6462	$ 84032
	DESIGN/ENGINEERING	28472	—	—	1	—	28473	
	DEVELOPMENT & QUAL. TEST HARDWARE	7025	2346	4066	161	5894	19492	
	FLIGHT TEST	1328	—	—	—	7130	8458	
	DEVELOPMENT TEST & QUALIFICATION	7776	68	855	24	—	8723	
	GRD. HDLG. EQUIP.	349	285	85	12	92	823	
	SUPPORT AND C/O EQ.	6797	199	59	11	241	7307	
	TOOLING	149	571	159	134	—	[illegible]	
	SPEC.TEST EQUIP.	827	1425	207	240	602	3301	
	SUBTOTAL	$ 57115	$ 5302	$ 5862	$ 631	$ 15122	$ 84032	
RECURRING PRODUCTION	PROGRAM MGT.	$ 395	$ 164	$ 165	$ 18	$ 474	$ 1216	$ 15814
	MANUFACTURING	—	1563	—	106	4912	6581	
	PROD. ASSUR.	—	—	468	—	—	468	
	ACCEPT. TEST	1357	68	1047	82	—	2554	
	TOOL & GSE MAINT.	325	99	20	16	37	497	
	SUSTAIN. ENGR.	2659	—	—	—	—	2659	
	SPARES	399	234	453	16	737	1839	
OPERATIONS	FLIGHT OPS. FLIGHT DATA REDUC.	$ 1644	—	—	—	—	$ 1644	$ 5349
	TRANSPORTATION	—	—		—	25	25	
	CHECKOUT	—	—	—	—	2100	2100	
	LAUNCH OPS.	185	—	—	—	—	185	
	SUST.ENG & PROG MGT	1395	—	—	—	—	1395	
	TOTAL	$ 65474	$ 7430	$ 8015	$ 869	$ 23407	$ 105195	$ 105195

* INCLUDES NON-ALLOCATED

Fig. 6-8

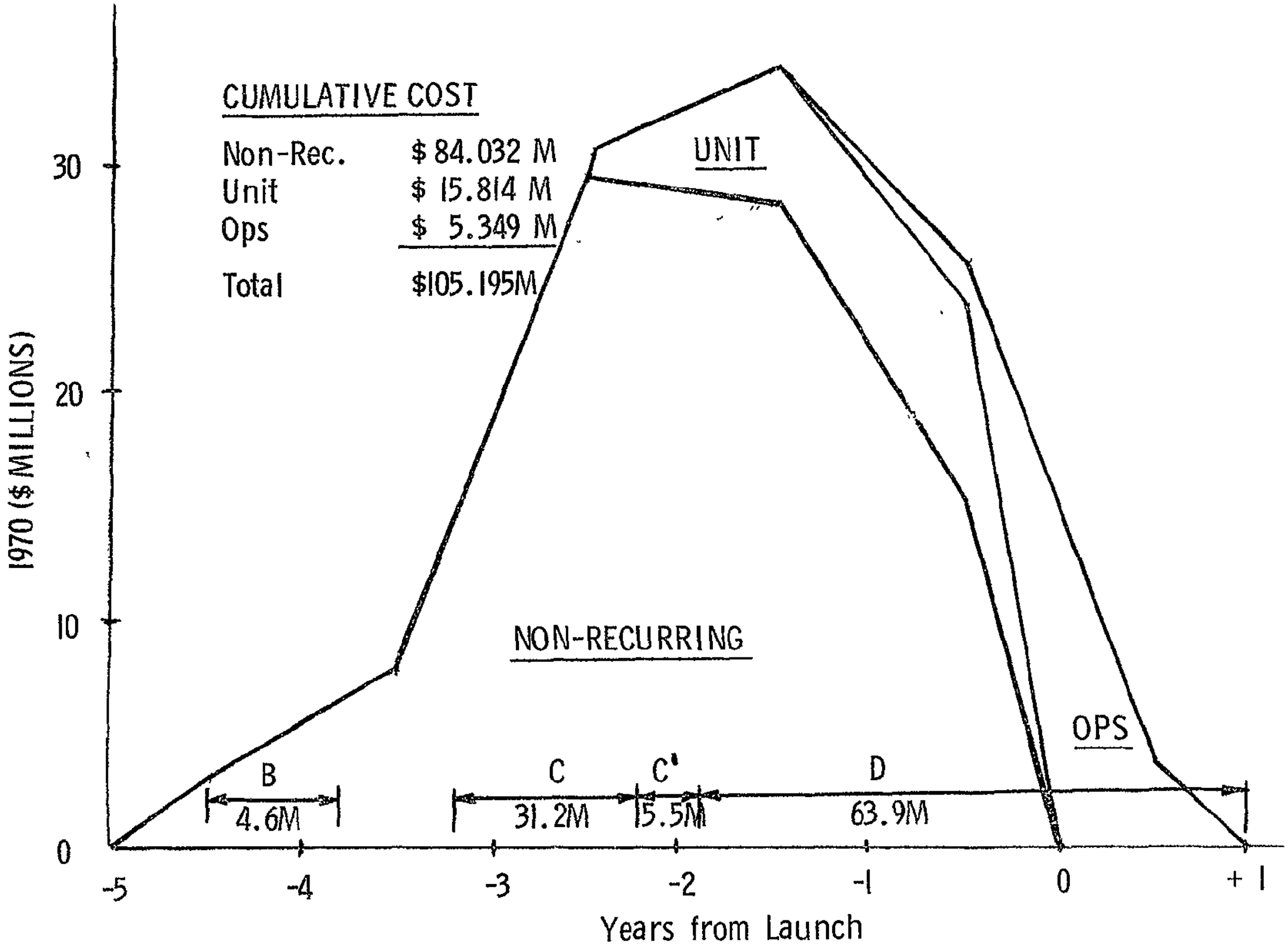
By Year

Year	Total	Non-Recurring	Recurring Unit	Recurring Operations
-5 (6 mos.)	$ 2948	$ 2948	-	-
-4	7684	7684	-	-
-3	30722	29639	$ 1083	-
-2	34651	28642	6009	-
-1	25802	15119	8722	1961
-1	3388	-	-	3388
Total	$105,195	$ 84032	$15,814	$ 5349

By Phase

B (9 mos.)	4,577	4,577	-	-
C (12 mos.)	31,221	30,679	542	-
C^1 (3 mos.)	5,556	5,015	541	-
D (36 mos.)	63,841	43,761	14,731	5349
Total	$105,195	$84,032	$15,814	$ 5349

Fig. 6-9 OAO-B Shuttle Funding (1970 $ Thousands)



LOW COST OAO-B ANNUAL FUNDING
(Space Shuttle-Launched)
Fig. 6-10

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6.4.1.2 Expendable-Launched OAO-B. The low-cost expendable (LCE) OAO-B cost estimates are summarized in Fig. 6-11 and 6-12. The RDT&E cost is estimated at $89.4M, the unit payload cost at $19.1M, and one year operations at $6.7M, resulting in a total program cost of $115.2M. This represents an increase of $10M over the Shuttle-launched OAO-B and is attributable to more extensive developmental testing, increased component redundancy, and related higher engineering design and sustaining costs. Product assurance costs are 38 percent higher than in the Shuttle case and engineering costs are 12 percent higher.

The time-phased funding for the low-cost expendable/alternate current expendable (LCE/ACE) OAO-B is shown in Figs. 6-13 and 6-14. The total program time span is the same as for the Shuttle case, with the annual funding peaking at $40M in year 2 prior to launch.

6.4.1.3 Recosted Baseline OAO-B. To assure a valid basis for comparison of low-cost and baseline OAO-B costs, the baseline OAO-B configuration was recosted. To maintain consistency in cost classification and provide a direct means of subsystem cost savings derivation, the same costing methodology, cost factors, rates, and assumptions were used to cost the historical baseline configuration as the low-cost derivatives.

The program was structured as closely as possible to the manner in which it was originally executed. Consistency with the cost structure was maintained. Actual subcontract and purchased equipment costs were gathered from the original OAO suppliers and subcontractors where possible.

Figure 6-15 shows costs for the recosted OAO-B baseline by subsystem. In total the costs are in agreement with the historical costs converted to 1970 dollars. However, there was considerable redistribution by cost category and subsystem due to the fact that original baseline costs were not collected at the subsystem level and had been allocated rather arbitrarily into non-recurring and recurring categories.

SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS
(1970 $ THOUSANDS)

PAYLOAD: OAO-B LAUNCH VEHICLE: LCE/ACE FLIGHT DURATION: 1 YEAR

SUBSYSTEM	COSTS			
	RDT&E COST	UNIT COST	UNIT OPERATIONS	TOTAL COST
PAYLOAD ASSEMBLY & INTEGRATION	$ 580	$ 141	$ 40	$ 761
EXPERIMENTS	10304	3075	1037	14416
STRUCTURES & MECHANISMS	7362	1050	338	8750
ELECTRICAL & PYROTECHNICS	11419	3401	760	15580
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL	29095	5902	1590	36587
PROPULSION & ATTITUDE CONTROL	4496	1100	180	5776
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)	21904	3671	2338	27913
ENVIRONMENTAL CONTROL	3050	445	166	3661
NON-ALLOCATED COSTS	1200	350	220	1770
PAYLOAD TOTAL	$ 89410	$ 19135	$ 6669	$ 115214

Fig. 6-11

DETAILED COST ALLOCATION TO EACH SUBSYSTEM
(1970 $ THOUSANDS)

SUBSYSTEM PAYLOAD SUMMARY* PAYLOAD TYPE OAO - B LCE / ACE

	COST CATEGORY	LABOR ENG.	LABOR MFG.	LABOR PA	MATERIEL MATERIAL	MATERIEL EQUIP/PARTS	SUB-TOTALS	TOTALS
RDT&E	PROGRAM MANAGEMENT	$ 4817	$ 465	$ 651	$ 54	$ 886	$ 6873	$ 89410
	DESIGN/ENGINEERING	31739	—	—	1	—	31740	
	DEVELOPMENT & QUAL. TEST HARDWARE	8120	2967	4925	185	7185	23382	
	FLIGHT TEST FACILITIES RENTAL	—	—	—	—	2135 400	2135 400	
	DEVELOPMENT TEST & QUALIFICATION	9796	68	2384	55	—	12303	
	GRD. HDLG. EQUIP.	371	305	91	13	98	878	
	SUPPORT AND C/O EQ.	6797	199	59	11	241	7307	
	TOOLING	174	621	153	143	—	1091	
	SPEC.TEST EQUIP.	827	1425	207	240	602	3301	
	SUBTOTAL	$ 62641	$ 6050	$ 8470	$ 702	$ 11547	$ 89410	
RECURRING PRODUCTION	PROGRAM MGT.	$ 470	$ 205	$ 199	$ 21	$ 576	$ 1471	$ 19135
	MANUFACTURING	—	1978	—	123	5988	8089	
	PROD. ASSUR.	—	—	591	—	—	591	
	ACCEPT. TEST	1442	68	1469	82	—	3061	
	TOOL. & GSE MAINT.	328	102	20	16	38	504	
	SUSTAIN. ENGR.	3181	—	—	—	—	3181	
	SPARES	693	307	309	31	898	2238	
OPERATIONS	FLIGHT OPS. FLIGHT DATA REDUC.	$ 1697	—	—	—	—	$ 1697	$ 6669
	TRANSPORTATION	—	—	—	—	30	30	
	CHECKOUT	—	—	—	—	2100	2100	
	LAUNCH OPS.	379	—	—	—	—	379	
	SUST.ENG & PROG MGT	2463	—	—	—	—	2463	
	TOTAL	$ 73294	$ 8710	$ 11058	$ 975	$ 21177	$ 115,214	$ 115,214

* INCLUDES NON-ALLOCATED COSTS.

Fig. 6-12

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By Year

Year	Total	Non-Recurring	Recurring Unit	Recurring Operations
-5	$ 3143 (6 mos.)	$ 3143	$ -	$ -
-4	8992	8992	-	-
-3	34765	33465	1300	-
-2	40284	32939	7345	-
-1	24155	10871	10490	2794
-1	3875	-	-	3875
Total	$115214	$ 89410	$ 19135	$ 6669

By Phase

	Total	Non-Recurring	Recurring Unit	Recurring Operations
B	5030 (9 mos.)	5030	-	-
C	35655 (12 mos.)	35005	650	-
C[1]	6215 (3 mos.)	5565	650	-
D	68314 (36 mos.)	43810	17835	6669
Total	$115214	$ 89410	$ 19135	$ 6669

Fig. 6-13 OAO-B LCE/ACE Funding (1970 $ Thousands)

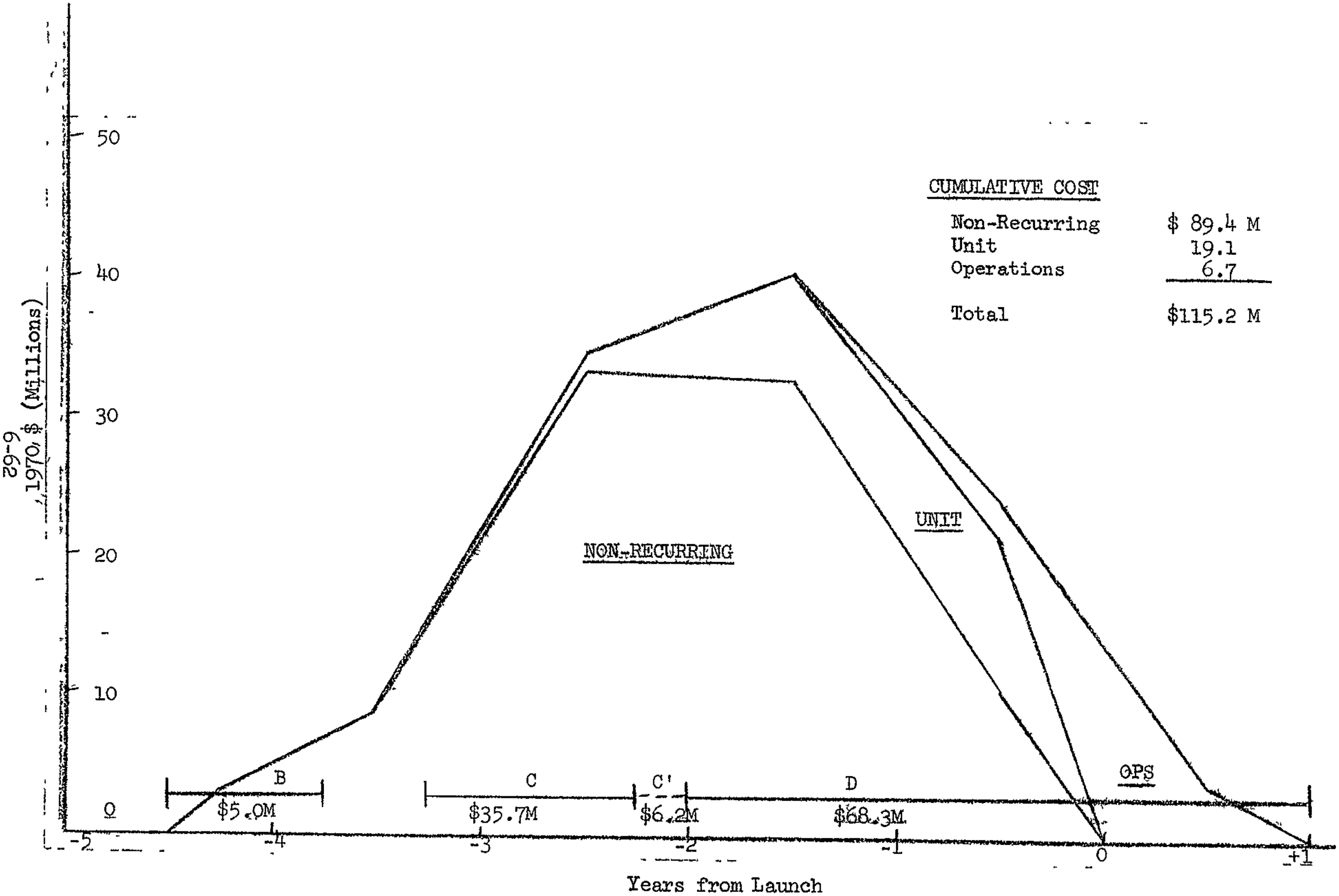

Fig. 6-14 Expendable Launched Low-Cost OAO-B Funding

SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS
(1970 $ THOUSANDS)

PAYLOAD: BASELINE OAO-B (RE-COSTED) LAUNCH VEHICLE: EXPENDABLE FLIGHT DURATION: 1 YR.

SUBSYSTEM	COSTS			
	RDT&E COST	UNIT COST	UNIT OPERATIONS	TOTAL COST
ADAPTER	$ 1088	$ 166	$ 81	$ 1335
EXPERIMENTS	15757	3573	2127	21457
STRUCTURES & MECHANISMS ASSY	11020	1156	393	12569
ELECTRICAL & PYROTECHNICS	17396	3583	726	21705
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL	72292	14302	3599	90193
PROPULSION & ATTITUDE CONTROL	4877	1074	209	6160
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)	38895	6921	3086	48902
ENVIRONMENTAL CONTROL	5028	973	298	6299
NON-ALLOCATED COSTS	1311	217	473	2001
PAYLOAD TOTAL	$ 167664	$ 31965	$ 10992	$ 210621 ✓

Fig. 6-15

The recosted baseline OAO-B was costed at the same level of detail as the low-cost OAO-B. The Payload Summary in Fig. 6-16 shows the functional cost aggregation, which results in $168M RDT&E cost, $32M unit cost, and $11M ooperations cost.

The recosted baseline OAO-B program costs were time phased over a 6-year period as shown in Fig. 6-17 and 6-18. A normal development time span was assumed, resulting in two peak funding years at over $60M during second and third years prior to launch.

6.4.1.4 OAO-B Program Cost Comparison. The results of OAO-B cost estimating are tabulated by subsystem in Fig. 6-19. Both the original baseline costs and the recosted OAO-B payload costs are shown for comparison with the Shuttle-launched and the LCE estimates.

The analysis of cost savings by subsystem and major cost category is shown in Fig. 6-20. These cost savings were derived using the recosted baseline and result in 45 percent and 50 percent savings in OAO-B program costs for the LCE and Shuttle-launched cases, respectively. The greatest savings appear in the Stabilization and Control (SCS), CDP&I (TT&C), and Environmental Control Subsystems. The SCS and CDP&I savings are due to use of fewer and simpler components permitted by the technology, and availability of greater volume and weight for the payload. The environmental control savings are due to use of passive techniques and thermal control of entire equipment section as opposed to individual modules. The 30 percent plus savings in structures, experiments, and electrical subsystems are primarily due to relaxed weight and volume constraints in the low-cost designs, resulting in decreased equipment density, and "ruggedized" manufacturing.

Another comparison of OAO-B costs is made in Fig. 6-21. Here the costs are compared by major cost category and element. The major savings in equipment/parts column bear out the component simplification and reduction conclusion. The material costs also show substantial savings due to use of cheaper, heavier

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DETAILED COST ALLOCATION TO EACH SUBSYSTEM

(1970 $ THOUSANDS)

SUBSYSTEM *PAYLOAD SUMMARY* * PAYLOAD TYPE *BASELINE OAO-B (RECOSTED)*

	COST CATEGORY	LABOR ENG.	LABOR MFG.	LABOR PA	MATERIEL MATERIAL	MATERIEL EQUIP/PARTS	SUB-TOTALS	TOTALS
RDT&E	PROGRAM MANAGEMENT	$ 6195	$ 769	$ 832	$ 120	$ 4977	$ 12893	
	DESIGN/ENGINEERING	41577	—	—	2	—	41579	
	DEVELOPMENT & QUAL. TEST HARDWARE	9309	4356	4618	829	54012	73124	
	FLIGHT TEST & FACILITIES **	—	—	—	—	—	—	
	DEVELOPMENT TEST & QUALIFICATION	19795	67	4090	80	—	24032	$ 167664
	GRD. HDLG. EQUIP.	581	331	101	16	130	1159	
	SUPPORT AND C/O EQ.	1468	2066	578	56	4376	8544	
	TOOLING	441	1748	422	406	—	3017	
	SPEC.TEST EQUIP.	1200	666	168	52	1230	3316	
	SUBTOTAL	$ 80566	$ 10003	$ 10809	$ 1561	$ 64725	$ 167664	
RECURRING PRODUCTION	PROGRAM MGT.	$ 657	$ 203	$ 155	$ 47	$ 1397	$ 2459	
	MANUFACTURING	—	1827	—	328	14383	16538	
	PROD. ASSUR.	—	—	546	—	—	546	$ 31965
	ACCEPT. TEST	1246	112	1027	145	—	2530	
	TOOL. & GSE MAINT.	148	192	51	21	229	641	
	SUSTAIN. ENGR.	5488	—	—	—	—	5488	
	SPARES	1010	291	236	71	2155	3763	
OPERATIONS	FLIGHT OPS. FLIGHT DATA REDUC.	$ 3173	—	—	—	—	$ 3173	
	TRANSPORTATION	—	—	—	—	30	30	
	CHECKOUT	—	—	—	—	2703	2703	$ 10992
	LAUNCH OPS.	961	—	—	—	—	961	
	SUST.ENG & PROG MGT	4125	—	—	—	—	4125	
	TOTAL	$ 97374	$ 12628	$ 12824	$ 2173	$ 85622	$ 210621	$ 210621 ✓

* INCLUDES NON-ALLOCATED COSTS. ** FACILITIES ARE GFE.

Fig. 6-16

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By Year

Year	Total	Non-Recurring	Recurring Unit	Recurring Operations
-5 (9 mos.)	$ 10,579	$ 10,579	$ -	$ -
-4	37,428	37,428	-	-
-3	68,448	64,783	3,665	-
-2	61,702	45,860	15,842	-
-1	26,159	9,014	12,458	4,687
-1	6,305	-	-	6,305
Total	$ 210,021	$ 167,664	$ 31,965	$ 10,992

By Phase

B (9 mos.)	$ 10,579	$ 10,579	-	-
C (12 mos.)	37,428	37,428	-	-
D (48 mos.)	162,614	119,657	31,965	10,992
Total	$ 210,621	$ 167,664	$ 31,965	$ 10,992

Fig. 6-17 Recosted OAO-B Funding (1970 $ Thousands)

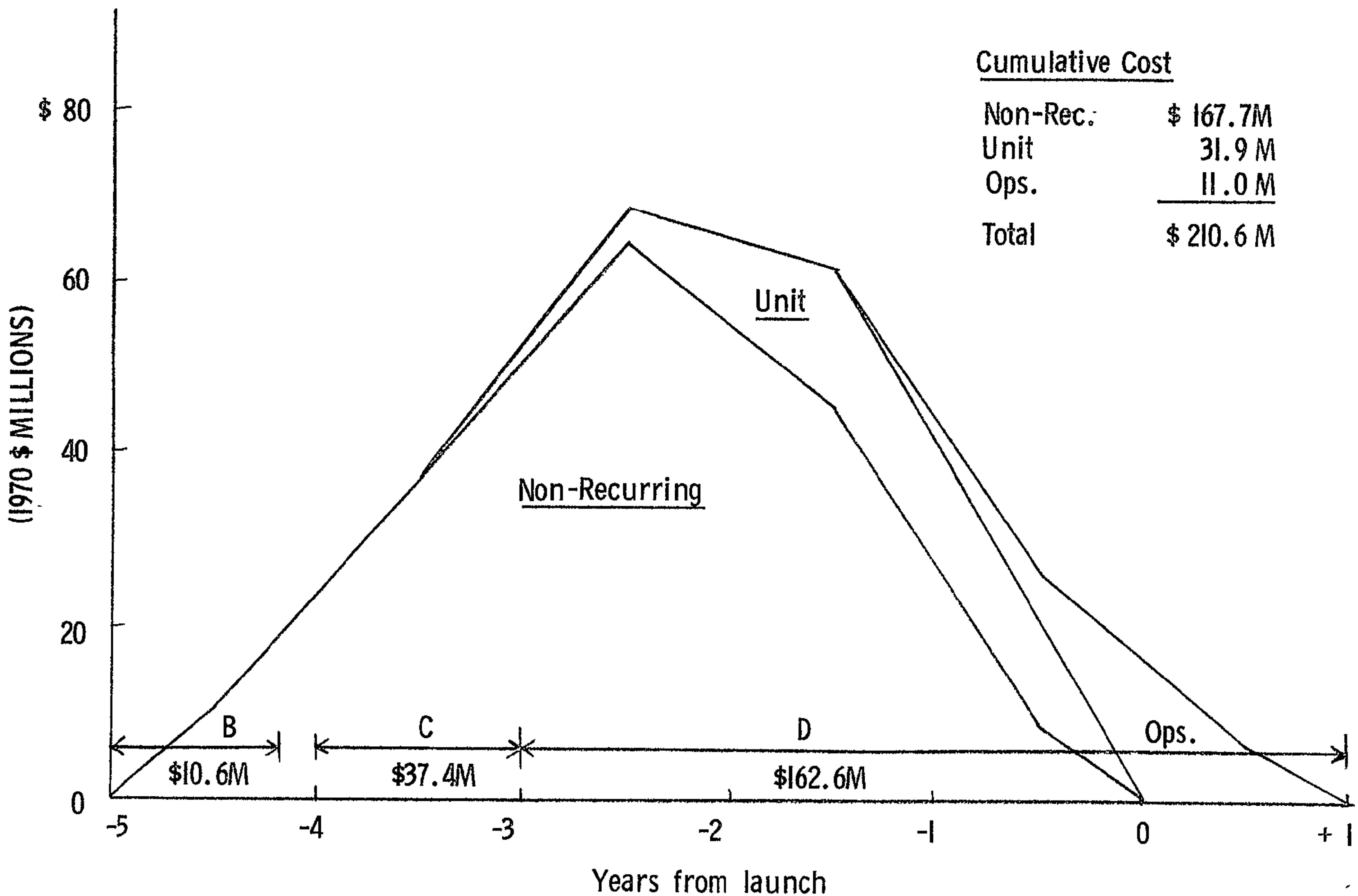

Fig. 6-18 Recosted Baseline OAO-B Annual Funding

OAO-B CDT COMPARISON

(1970 $ MILLIONS)

SUBSYSTEM	ADJUSTED BASELINE COSTS			RECOSTED BASELINE OAO-B			SHUTTLE LAUNCHED OAO-B			EXPENDABLE LAUNCHE OAO-B		
	RDT&E	UNIT	OPS.	RDT&E	UNIT	OPS.	RDT&E	UNIT	OPS.	RDT&E	UNIT	OPS
ADAPTER	$ 0.60	$ 0.15	$ 0.10	$ 1.09	$ 0.17	$ 0.08	—	—	—	$ 0.58	$ 0.14	$ 0.04
EXPERIMENTS	11.52	5.00	2.20	15.76	3.57	2.13	$ 9.91	$ 2.91	$ 0.80	10.30	3.08	1.04
STRUCTURE & MECHANICAL	9.04	5.10	0.40	11.02	1.16	0.39	6.69	0.95	0.15	7.36	1.05	0.34
ELECTRICAL	17.08	3.90	0.76	17.40	3.58	0.73	11.24	2.81	0.72	11.42	3.40	0.76
STABILIZATION & CONTROL	76.65	11.50	3.90	72.29	14.30	3.60	27.00	4.67	1.19	29.10	5.90	1.59
ATTITUDE CONTROL	5.10	1.50	0.20	4.88	1.07	0.21	4.30	1.01	0.18	4.50	1.10	0.18
CDPI	40.82	4.60	2.80	38.89	6.92	3.09	20.90	2.75	2.13	21.90	3.67	2.34
ENVIRONMENTAL CONTROL	6.05	1.00	0.55	5.03	0.97	0.30	2.90	0.39	0.07	3.05	0.45	0.17
UNALLOCATED	1.35	0.60	0.31	1.31	0.22	0.47	1.10	0.33	0.12	1.20	0.35	0.22
TOTAL	$ 168.21	$ 33.35	$ 11.22	$ 167.67	$ 31.96	$ 11.00	$ 84.04	$ 15.82	$ 5.36	$ 89.41	$ 19.14	$ 6.68

Fig. 6-19

Subsystem	RECOSTED BASELINE VS LCE				RECOSTED BASELINE VS SHUTTLE			
	RDT&E	Unit	Ops.	Total	RDT&E	Unit	Ops.	Total
Experiments	35%	14%	51%	33%	37%	18%	62%	37%
Struct. & Mech.	33	9	13	30	39	18	62	38
Electrical	34	5	(-4)	28	35	22	1	32
Stab. & Control	60	59	56	59	63	67	67	64
Attitude Control	8	(-3)	14	6	12	6	14	11
CDPI	44	47	24	43	46	60	31	47
Env. Control	39	54	43	42	42	60	77	47
Total*	47%	40%	39%	45%	50%	51%	51%	50%

* Adapter & Non-Allocated costs included.

Fig. 6-20 % Cost Savings by Subsystem for OAO-B

COST	CONFIGURATION	LABOR			MATERIAL		TOTAL COST
		Eng.	Mfg.	P.A.	Material	Equip/Parts	
RDT&E	Baseline LCE Shuttle	$ 80566 62641 57115	$ 10003 6050 5302	$ 10809 8470 5862	$ 1561 702 631	$ 64725 11547 15122*	$ 167664 89410 84032
% Savings	LCE Shuttle	22% 29%	40% 47%	22% 46%	55% 60%	82% 77%	47% 50%
Unit	Baseline LCE Shuttle	8549 6114 5135	2625 2660 2128	2015 2588 2153	612 273 238	18164 7500 6160	31965 19135 15814
% Savings	LCE Shuttle	28% 40%	-1% 19%	-28% - 7%	55% 61%	59% 66%	40% 51%
Operations	Baseline LCE Shuttle	8259 4539 3224	- - -	- - -	- - -	2733 2130 2125	10992 6669 5349
% Savings	LCE Shuttle	45% 61%	-	-	-	22% 22%	39% 51%
Total	Baseline LCE Shuttle	97374 73294 65474	12628 8710 7430	12824 11058 8015	2173 975 869	85622 21177 23407*	210621 115214 105195
% Savings	LCE Shuttle	25% 33%	31% 41%	14% 37%	55% 60%	75% 73%	45% 50%

Fig. 6-21 OAO-B Functional Cost Comparison & % Savings (1970 $ Thousands)

*Note: Includes Shuttle Sortie Flight Test at $5 M.

materials. The negative savings shown in the unit cost product assurance (PA) are due to more extensive testing at the unit/production levels as opposed to the R&D developmental testing (i.e., eliminated structural test), and the effect of module level testing (particularly for the expendable case). In the Shuttle case, PA is relaxed, but still higher than in the baseline since module are utilized. The negative savings effect of module manufacturing appears unde LCE unit manufacturing, where higher modular construction cost slightly overcor the weight/volume savings.

Further in-depth comparisons and cost savings analysis for the payloads studies are treated in Section 6.5.

6.4.2 SEO Cost Estimates

The SEO payload is a Lunar Orbiter derivative including as its major experiment the Lunar Orbiter photo camera package and two additional AVCS TV cameras. The basic SEO program consists of four spacecraft in orbit for a two-year mission period plus one additional spacecraft as the backup vehicle.

6.4.2.1 Shuttle-Launched SEO. The costs by subsystem for the Shuttle-launched SEO are presented in Fig. 6-22. Payload development cost is shown, followed by an average unit cost, then two year operations costs for four on-orbit spacecraft, and the total program cost, which includes five spacecraft and two sets of non-redundant spare modules in support of the SEO program.

The experiment subsystem RDT&E and unit costs are shown broken down for the photo package subcontract, the AVCS camera, and the prime contractors integration costs. In the payload summary, Fig. 6-23, the AVCS development cost of $12M is carried as a single entry, since no functional breakdown was available for this cost obtained from NASA.

The SEO functional cost summary shows the RDT&E cost breakdown, the single unit cost breakdown, and the two-year operations cost for four units. Note that the total column in Fig. 6-23 does not represent the total program cost of Fig. 6-2

SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS
(1970 $ THOUSANDS)

PAYLOAD: SEO LAUNCH VEHICLE: SHUTTLE FLIGHT DURATION: 2 YRS.

SUBSYSTEM		COSTS			
		RDT&E COST	UNIT COST	4 UNITS/2 YRS OPERATIONS	TOTAL COST *
ADAPTER ~~LAUNCH VEHICLE~~		—	—	—	—
~~ADAPTER~~	PHOTO SUBCONTR.	$ 20,003	$ 1280	$ 2930	$ 55,276
EXPERIMENTS	EXPER. PRIME	5535	171	1751	
	AVCS	12 000	491		
STRUCTURES & MECHANISMS		6047	539	740	9482
ELECTRICAL & PYROTECHNICS		10208	1716	1574	22309
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL		9543	1862	1910	24006
PROPULSION & ATTITUDE CONTROL		3361	638	507	7337
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)		17761	2794	2876	37716
ENVIRONMENTAL CONTROL		837	129	168	1650
NON-ALLOCATED COSTS		405	210	312	1767
PAYLOAD TOTAL		$ 85700	$ 9830	$ 12768	$ 159543

* INCLUDES 5 SPACECRAFT UNITS PLUS TWO SETS OF NON-REDUNDANT MODULES AND OPERATIONS FOR FOUR FLIGHT UNITS OVER A TWO YEAR PERIOD.

Fig. 6-22

DETAILED COST ALLOCATION TO EACH SUBSYSTEM
(1970 $ THOUSANDS)

SUBSYSTEM PAYLOAD SUMMARY * PAYLOAD TYPE SEO / SHUTTLE

	COST CATEGORY	LABOR			MATERIEL		SUB-TOTALS	TOTALS
		ENG.	MFG.	PA	MATERIAL	EQUIP/PARTS		
RDT&E	PROGRAM MANAGEMENT	$ 3335	$ 464	$ 446	$ 80	$ 1342	$ 5667	$ 85,700
	DESIGN/ENGINEERING	24591	—	—	1	—	24592	
	DEVELOPMENT & QUAL. TEST HARDWARE	4869	1646	2958	224	4644	14341	
	~~[illegible]~~ ** AVCS DEVELOPMENT	12000	—	—	—	—	12000	
	DEVELOPMENT TEST & QUALIFICATION	5610	42	1311	76	—	7039	
	GRD. HDLG. EQUIP.	454	389	239	58	63	1203	
	SUPPORT AND C/O EQ.	2544	1253	389	147	10093	14426	
	TOOLING	328	857	232	99	—	1516	
	SPEC.TEST EQUIP.	1640	1392	223	354	1307	4916	
	SUBTOTAL	$ 55371	$ 6043	$ 5798	$ 1039	$ 17449	$ 85700	
RECURRING PRODUCTION	PROGRAM MGT.	$ 208	$ 105	$ 120	$ 20	$ 303	$ 756	$ 9830
	MANUFACTURING	—	1033	—	127	3075	4235	
	PROD. ASSUR.	—	—	309	—	—	309	
	ACCEPT. TEST	652	34	931	79	—	1696	
	TOOL. & GSE MAINT.	45	35	10	6	102	198	
	SUSTAIN. ENGR.	1479	—	—	—	—	1479	
	SPARES	320	160	186	30	461	1157	
OPERATIONS	FLIGHT OPS. FLIGHT DATA REDUC.	$ 1995	—	—	—	—	$ 1995	$ 12768
	TRANSPORTATION	—	—	—	—	125	125	
	CHECKOUT	—	—	—	—	3200	3200	
	LAUNCH OPS.	1488	—	—	—	—	1488	
	SUST.ENG & PROG MGT	5960	—	—	—	—	5960	
	TOTAL	$ 67518	$ 7410	$ 7354	$ 1301	$ 24715	$ 108298	$ 108298

* INCLUDES NON-ALLOCATED COSTS. ** AVCS TOTAL DEVELOPMENT COST.

Fig. 6-23

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This is because the data in Fig. 6-22 includes 5 payloads and spare modules. The Shuttle-launched SEO total program cost of $159.6M was time-phased over a $5\frac{1}{2}$-year time span as shown in Fig. 6-24 and plotted in Fig. 6-25.

6.4.2.2 Expendable-Launched SEO. Figures 6-26 and 6-27 present the subsystem cost breakdown and the functional cost breakdown for the LCE SEO, respectively. The total program cost for the expendable-launched SEO includes 5 spacecraft, four two-year unit operations, and development cost, resulting in $168.2M total. Spare non-redundant modules are not included in the expendable case. The experiment subsystem is similarly treated as for the SEO/Shuttle case and the functional cost breakdown is also in the same format.

The expendable-launched SEO funding is spread over a 6 year time period including the two operating years. Figures 6-28 and 6-29 show the time-phased cost tabulation and plot, respectively.

6.4.2.3 Recosted Baseline SEO. Since the SEO is a synthesized payload derived from the Lunar Orbiter, the original baseline SEO cost estimates were also synthesized from the Lunar Orbiter costs and use of cost estimating relationships. The Lunar Orbiter was a short mission duration spacecraft in relation to the two year SEO, thus with the addition of redundant components and mission extension to two years, the synthesized SEO baseline costs needed to be reexamined.

The baseline SEO configuration was recosted in detail using the "bottom up" pricing technique, same as for the baseline OAO-B and the low-cost payloads. The resulting cost estimates by subsystem are shown in Fig. 6-30. The total program cost of $202.6M is based on 5 spacecraft and two-year operations for four on-orbit spacecraft and the $120.8M development cost. The experiment subsystem is treated in the same manner as in the low-cost SEO.

Figure 6-31 provides the summary functional cost breakdown of the recosted baseline SEO RDT&E, unit, and operations costs. The total in this figure is shown for cross-checking purposes only, since it does not include the other four spacecraft needed by the total program.

By Year

Year	Total	Non-Recurring	Recurring Units	Recurring Operations
-4 (6 mos)	5159	5159	-	-
-3 (6 mos)	20624	18782	1842	-
-2	58083	46741	11342	-
-1	56687	14476	39700	2511
+1	14884	542	8191	6151
+2	4106	-	-	4106
Total	$ 159543	$ 85700	$ 61075	$ 12768

By Phase

B (6 mos)	5159	5159	-	-
C (12 mos)	48494	42860	5634	-
D (42 mos)	105890	37681	55441	12768
Total	$ 159543	$ 85700	$ 61075	$ 12768

Fig. 6-24 SEO/Shuttle Funding (1970 $ Thousands)

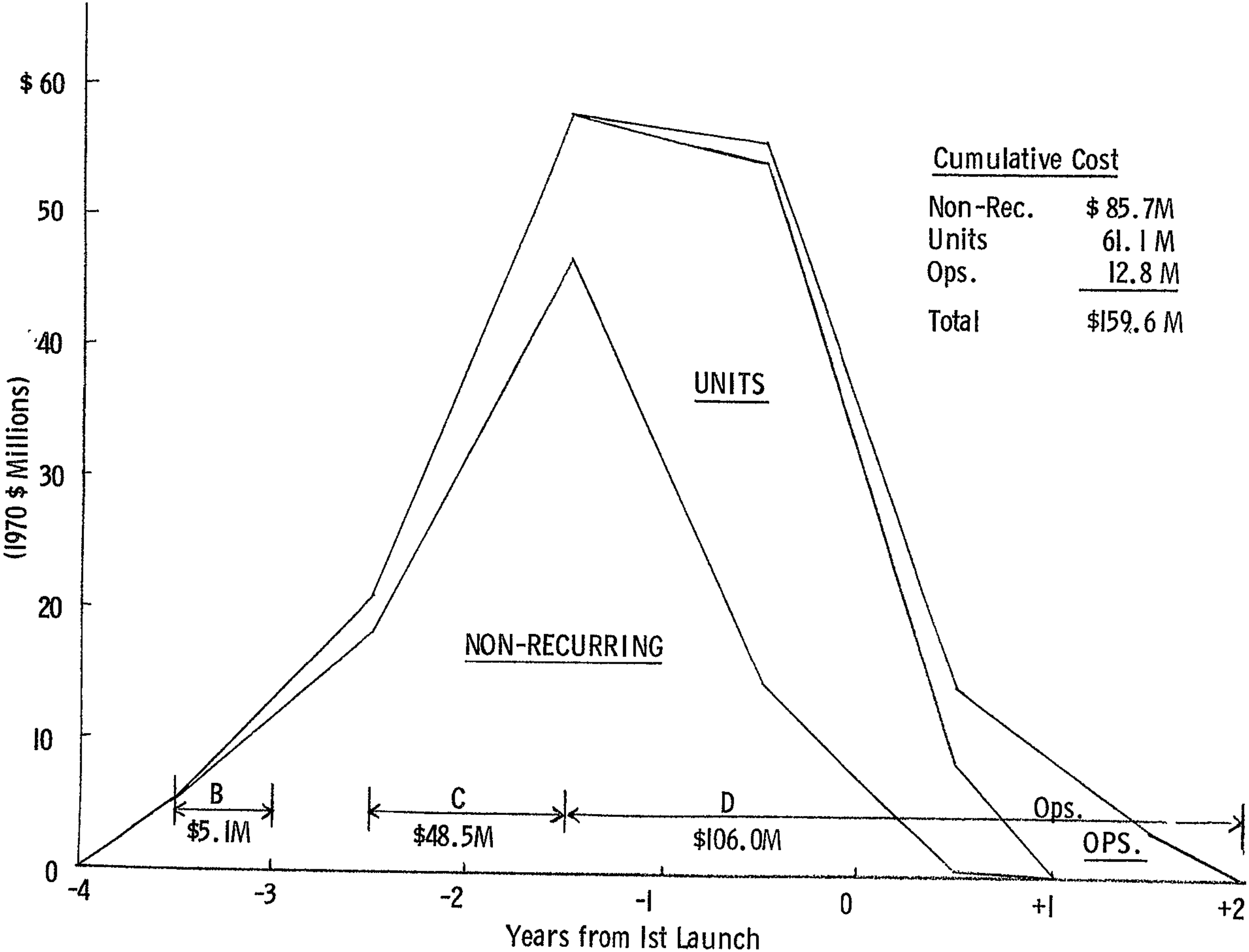

Fig. 6-25 Low Cost SEO Annual Funding (Space Shuttle-Launched)

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SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS

(1970 $ THOUSANDS)

PAYLOAD: SEO LAUNCH VEHICLE: LCE / ACE FLIGHT DURATION: 2 YRS.

SUBSYSTEM		COSTS			
		RDT&E COST	UNIT COST	4 UNITS / 2 YRS. OPERATIONS	TOTAL COST *
~~LAUNCH VEHICLE~~ ADAPTER		$ 926	$ 134	$ 97	$ 1693
~~ADAPTER~~ EXPERIMENTS	PHOTO SUBCONTR.	22642	1436	2930	56645
	EXPER. PRIME	6068	209	2160	
	AVCS	12000	524		
STRUCTURES & MECHANISMS		6523	538	810	10023
ELECTRICAL & PYROTECHNICS		11762	1928	1661	23063
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL		11117	1973	2292	23274
PROPULSION & ATTITUDE CONTROL		3890	727	525	8050
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)		21387	3328	3312	41339
ENVIRONMENTAL CONTROL		1002	129	171	1818
NON-ALLOCATED COSTS		672	243	419	2306
PAYLOAD TOTAL		$ 97989	$ 11169	$ 14377	$ 168211

* INCLUDES 5 SPACECRAFT AND OPERATIONS FOR FOUR FLIGHT UNITS OVER A TWO YEAR PERIOD.

Fig. 6-26

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DETAILED COST ALLOCATION TO EACH SUBSYSTEM

(1970 $ THOUSANDS)

SUBSYSTEM PAYLOAD SUMMARY * PAYLOAD TYPE SEO / LCE & ACE

	COST CATEGORY	LABOR ENG.	LABOR MFG.	LABOR PA	MATERIEL MATERIAL	MATERIEL EQUIP/PARTS	SUB-TOTALS	TOTALS
RDT&E	PROGRAM MANAGEMENT	$ 3757	$ 591	$ 583	$ 89	$ 1592	$ 6612	$ 97989
	DESIGN/ENGINEERING	27457	—	—	1	—	27458	
	DEVELOPMENT & QUAL. TEST HARDWARE	6933	3076	4566	330	7646	22551	
	~~FLIGHT TEST~~ AVCS DEVELOPMENT	12000 **	—	—	—	—	12000	
	DEVELOPMENT TEST & QUALIFICATION	5710	65	1329	77	—	7181	
	GRD. HDLG. EQUIP.	476	409	245	59	69	1258	
	SUPPORT AND C/O EQ.	2544	1253	389	147	10093	14426	
	TOOLING	344	896	243	104	—	1587	
	SPEC.TEST EQUIP.	1640	1392	223	354	1307	4916	
	SUBTOTAL	$ 60861	$ 7682	$ 7578	$ 1161	$ 20707	$ 97989	
RECURRING PRODUCTION	PROGRAM MGT.	$ 228	$ 112	$ 171	$ 21	$ 327	$ 859	$ 11169
	MANUFACTURING	—	1100	—	130	3330	4560	
	PROD. ASSUR.	—	—	329	—	—	329	
	ACCEPT. TEST	816	35	1445	84	—	2380	
	TOOL. & GSE MAINT.	46	35	10	6	102	199	
	SUSTAIN. ENGR.	1523	—	—	—	—	1523	
	SPARES	351	170	266	32	500	1319	
OPERATIONS	FLIGHT OPS. FLIGHT DATA REDUC.	$ 1995	—	—	—	—	$ 1995	$ 14377
	TRANSPORTATION	—	—	—	—	150	150	
	CHECKOUT	—	—	—	—	3200	3200	
	LAUNCH OPS.	3050	—	—	—	—	3050	
	SUST.ENG & PROG MGT	5982	—	—	—	—	5982	
	TOTAL	$ 74852	$ 9134	$ 9799	$ 1434	$ 28316	$ 123535	$ 123535

* INCLUDES NON-ALLOCATED COSTS

** AVCS TOTAL DEVELOPMENT COST.

Fig. 6-2

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By Year

Year	Total	Non-Recurring	Recurring Units	Recurring Operations
-4 (6 mos)	$ 4838	$ 4838	$ -	$ -
-3	41689	40172	1517	-
-2	52299	41228	10671	400
-1	49785	11209	35413	3163
+1	15494	542	8244	6708
+2	4106	-	-	4106
Total	$ 168211	$ 97989	$ 55845	$ 14377

By Phase

B (6 mos)	4838	4838	-	-
C (12 mos)	41689	40172	1517	-
D (48 mos)	121684	52979	54328	14377
Total	$ 168211	$ 97989	$ 55845	$ 14377

Fig. 6-28 SEO ACE/LCE Funding (1970 $ Thousands)

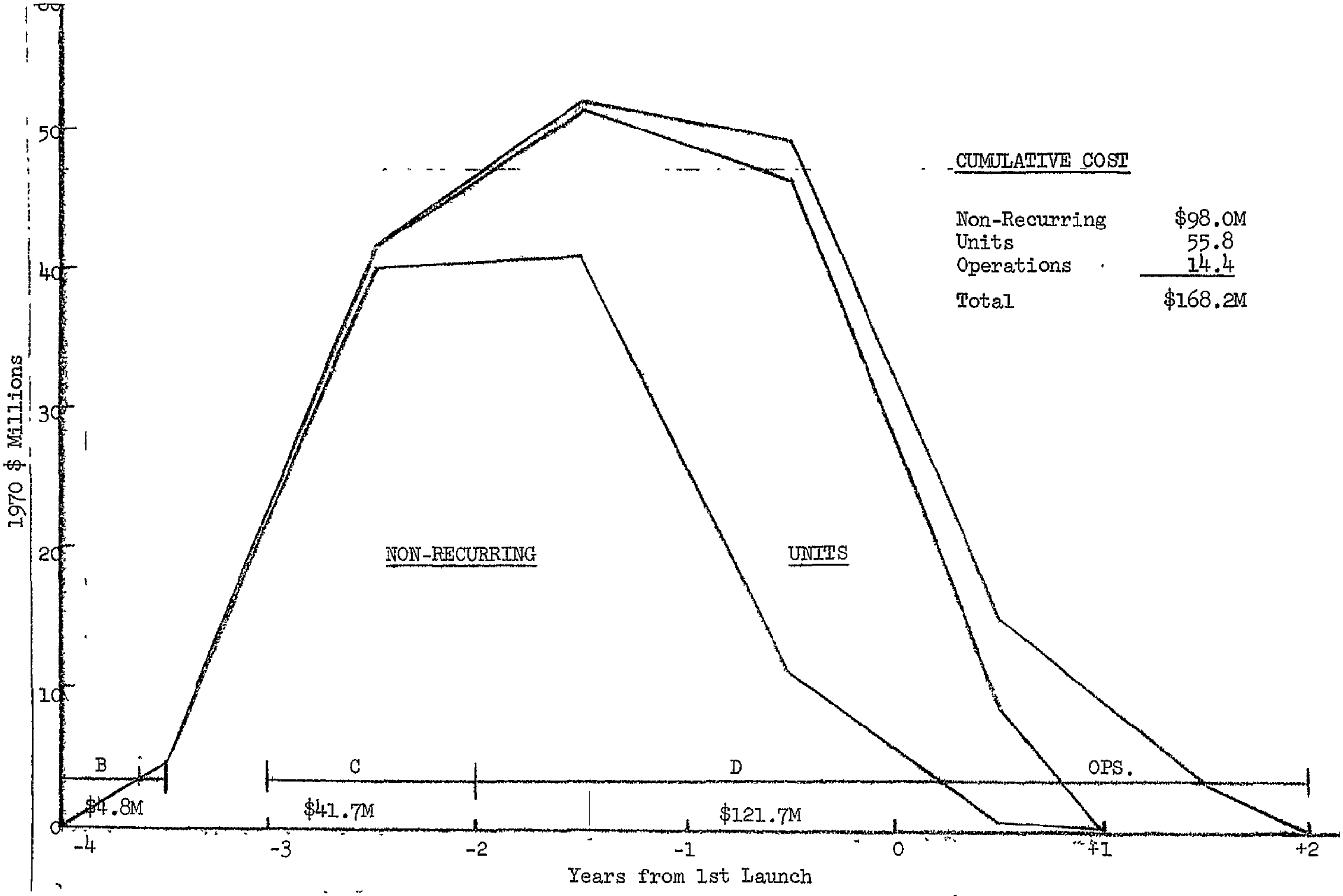

Fig. 6-29 Low Cost SEO Annual Funding (Expendable-Launched)

SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS

(1970 $ THOUSANDS)

PAYLOAD: BASELINE SEO (RECOSTED) LAUNCH VEHICLE: EXPENDABLE FLIGHT DURATION: 2 YRS.

SUBSYSTEM	COSTS			
	RDT&E COST	UNIT COST	4 UNIT 2 YR. OPERATIONS	TOTAL PROGRAM COST *
ADAPTER ~~PAYLOAD ASSEMBLY~~	$ 1247	$ 199	$ 97	$ 2339
~~& INTEGRATION~~ PHOTO SUBCONTR	27929	2160	3362	
EXPERIMENTS: EXPER. PRIME	6947			
EXPERIMENTS: AVCS	12000	856	2210	67528
STRUCTURES & MECHANISMS	10594	1264	934	17848
ELECTRICAL & PYROTECHNICS	14090	2056	1993	26363
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL	17173	2492	2633	32266
PROPULSION & ATTITUDE CONTROL	4076	549	546	7367
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)	25394	3335	3722	45791
ENVIRONMENTAL CONTROL	1110	153	214	2089
NON-ALLOCATED COSTS	265	88	342	1047
PAYLOAD TOTAL	$ 120825	$ 13152	$ 16053	$ 202638

* INCLUDES 5 SPACECRAFT AND TWO YEAR OPERATIONS FOR FOUR ON ORBIT SPACECRAFT.

Fig. 6-20

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DETAILED COST ALLOCATION TO EACH SUBSYSTEM
(1970 $ THOUSANDS)

SUBSYSTEM PAYLOAD SUMMARY * PAYLOAD TYPE BASELINE SEO (RECOSTED)

	COST CATEGORY	LABOR			MATERIEL		SUB-TOTALS	TOTALS
		ENG.	MFG.	PA	MATERIAL	EQUIP/PARTS		
RDT&E	PROGRAM MANAGEMENT	$ 4347	$ 1084	$ 879	$ 98	$ 1960	$ 8368	
	DESIGN/ENGINEERING	33230	—	—	1	—	33231	
	DEVELOPMENT & QUAL. TEST HARDWARE	5029	7094	6920	336	10951	30330	
	~~FLIGHT TEST~~ AVCS DEVELOPMENT	** 12000	—	—	—	—	12000	$ 120825
	DEVELOPMENT TEST & QUALIFICATION	8146	110	2027	78	—	10361	
	GRD. HDLG. EQUIP.	562	474	273	61	78	1448	
	SUPPORT AND C/O EQ.	3014	2451	733	158	11095	17451	
	TOOLING	476	1439	369	167	—	2451	
	SPEC.TEST EQUIP.	1726	1450	234	373	1402	5185	
	SUBTOTAL	$ 68530	$ 14102	$ 11435	$ 1272	$ 25486	$ 120825	
RECURRING PRODUCTION / UNIT	PROGRAM MGT.	$ 259	$ 203	$ 192	$ 22	$ 335	$ 1011	
	MANUFACTURING	—	2022	—	110	3395	5527	
	PROD. ASSUR.	—	—	604	—	—	604	
	ACCEPT. TEST	874	44	1381	118	—	2417	$ 13152
	TOOL. & GSE MAINT.	58	58	16	7	125	264	
	SUSTAIN. ENGR.	1778	—	—	—	—	1778	
	SPARES	398	310	298	35	510	1551	
OPERATIONS 4 UNIT 2 YRS	FLIGHT OPS. FLIGHT DATA REDUC.	$ 1995	—	—	—	—	$ 1995	
	TRANSPORTATION	—	—	—	—	150	150	
	CHECKOUT	—	—	—	—	4154	4154	$ 16053
	LAUNCH OPS.	2798	—	—	—	—	2798	
	SUST.ENG & PROG MGT	6956	—	—	—	—	6956	
	TOTAL	$ 83646	$ 16739	$ 13926	$ 1564	$ 34155	$ 150030	$ 150030

*INCLUDES NON-ALLOCATED COSTS. **TOTAL AVCS DEVELOPMENT COST.

Fig. 6-

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The recosted baseline SEO was funded over a 6-year period, 4 years developmental and 2 years of operations. Figures 6-32 and 6-33 show the annual funding required and the plot thereof.

6.4.2.4 SEO Cost Comparison. The SEO RDT&E, unit, and operations cost estimates for the cases studied are summarized for comparison purposes in Fig. 6-34. The original baseline costs are shown for reference only, since all savings comparisons were made with respect to the recosted baseline SEO.

Figure 6-35 shows the percentage cost savings by subsystem for the RDT&E, unit, operations, and total program cost. At the total level, savings in costs are 21 percent for the Shuttle-launched SEO and 17 percent for the LCE SEO. At the individual subsystem level, the largest savings appear in the structures subsystem, followed by the Stabilization & Control Subsystem. The cost reduction in structures is due to use of low-cost materials and simple, rectangular box design with drawer-like modules. The structure is fabricated from commercial grade aluminum, not requiring elaborate machining and milling during manufacturing, nor any structural testing. The SCS subsystem savings result from less elaborate electronics, such as combining two flight electronics control assemblies into a single package and removing the Polaris star trackers. The simpler, rigidized, electronics require considerably less testing and PA support effort as well as simpler manufacturing.

The negative savings in the LCE attitude control subsystem result from use of a large subsystem to control the much heavier (172 percent) spacecraft. In the expendable case this subsystem is subjected to expensive unit acceptance testing, which is not overcome by the low-cost manufacturing technique savings. The 10 to 20 percent savings in the other subsystems result from the low-cost design, manufacturing, and test approaches used in the low-cost SEO programs.

Functional cost comparison of the SEO is shown in Fig. 6-36. The comparison is made for the RDT&E cost, the single unit cost, four unit two year operations costs, and the total of the above by major cost element. As shown, the

By Year

Year	Total	Non-Recurring	Recurring Units	Recurring Operations
-4	$ .7352	$ 7352	$ -	$ -
-3	47367	45850	1517	-
-2	62139	50299	11320	520
-1	62579	16782	42462	3335
+1	18521	542	10461	7518
+2	4680	-	-	4680
Total	$202638	$120825	$65760	$16053

By Phase

B (6 mos)	5517	5517	-	-
Photo Sub. Design Cont.	1835	1835	-	-
C (12 mos)	47367	45850	1517	-
D (48 mos)	147919	67623	64243	16053
Total	$202638	$120825	$65760	$16053

Fig. 6-32 Recosted Baseline SEO Funding (1970 $ Thousands)

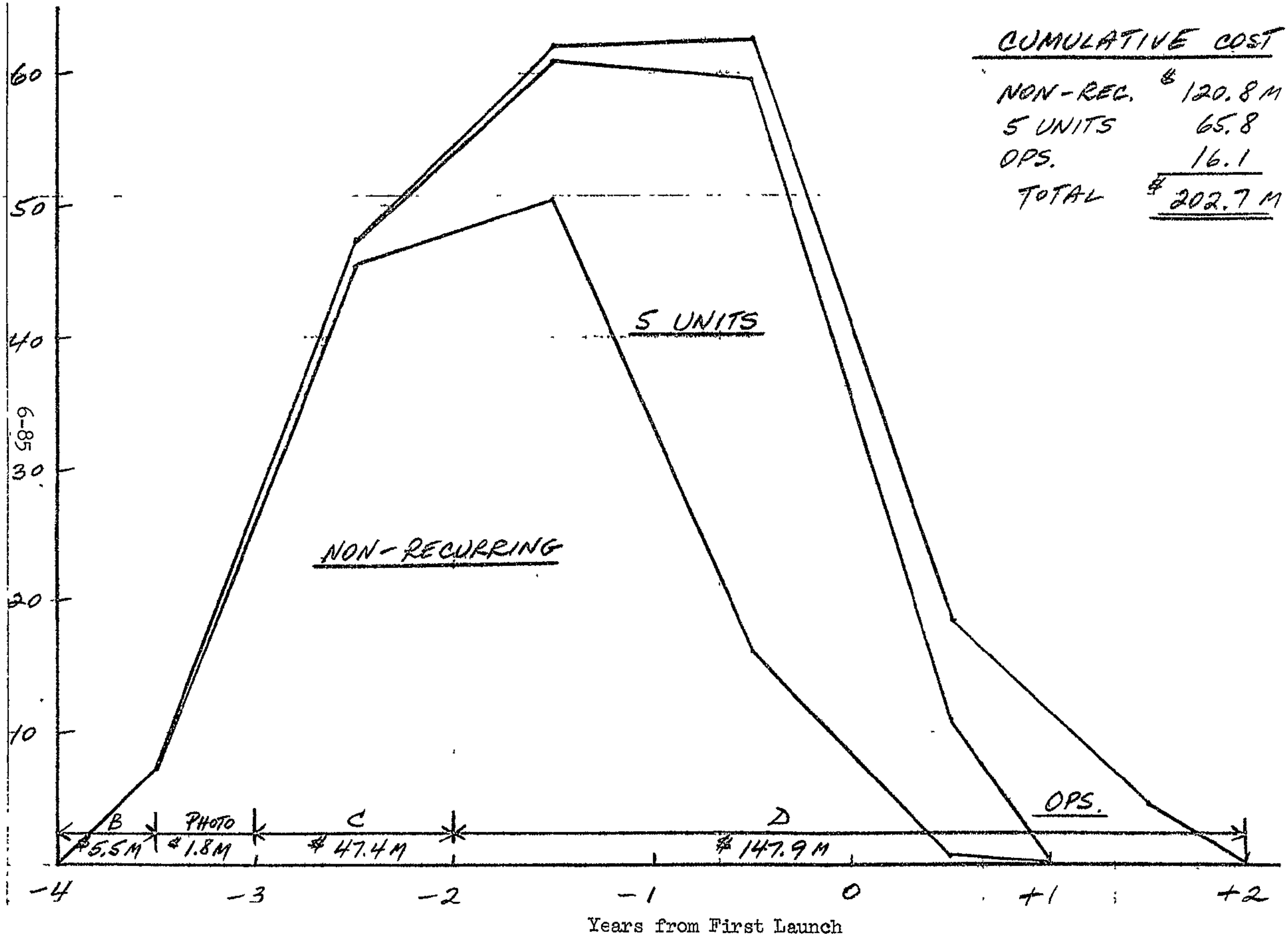

Fig. 6-33 Recosted Baseline SEO Funding

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SUB-SYSTEM	ADJUSTED BASELINE COSTS				RECOSTED BASELINE SEO				SHUTTLE LAUNCHED SEO				EXPENDABLE LAUNCHED SEO			
	RDT&E	UNIT	4 UNIT OPS.	TOT. PROGR.*	RDT&E	UNIT	4 UNIT OPS.	TOT. PROGR.*	RDT&E	UNIT	4** UNIT OPS.	TOT. PROGR.*	RDT&E	UNIT	4** UNIT OPS.	TOT. PROGR.*
ADAPTER	$ 2.9	$ 0.2	$ 1.7	$ 5.7	$ 1.2	$ 0.2	$ 0.1	$ 2.3	—	—	—	—	$ 0.9	$ 0.2	$ 0.1	$ 1.7
EXPERIMENT	47.1	2.5	9.3	69.1	46.9	3.0	5.6	67.5	$ 37.5	1.9	$ 4.7	$ 55.3	40.7	2.2	5.1	56.7
STRUCTURE	6.9	1.2	0.8	13.6	10.6	1.3	0.9	17.9	6.0	0.6	0.7	9.5	6.5	0.6	0.8	10.0
ELECTRIC.	12.4	1.9	1.9	23.9	14.1	2.1	2.0	26.4	10.2	1.7	1.6	22.3	11.8	1.9	1.7	23.1
SCS	15.9	3.3	2.5	34.8	17.2	2.5	2.7	32.3	9.6	1.9	1.9	24.0	11.1	2.0	2.3	23.3
ACS	4.0	0.6	0.5	7.8	4.1	0.6	0.6	7.4	3.4	0.6	0.5	7.3	3.9	0.7	0.5	8.0
CDPI	25.9	3.9	4.3	49.6	25.4	3.3	3.7	45.8	17.8	2.8	2.9	37.7	21.4	3.3	3.3	41.3
ECS	0.5	0.1	0.1	0.7	1.1	0.1	0.2	2.1	0.8	0.1	0.2	1.7	1.0	0.1	0.2	1.8
UNALLOC.	1.1	0.2	1.2	3.3	0.3	0.1	0.3	1.0	0.4	0.2	0.3	1.8	0.7	0.2	0.4	2.3
TOTAL*	$ 116.7	$ 13.9	$ 22.3	$ 208.5	$ 120.8	$ 13.2	$ 16.1	$ 202.7	$ 85.7	$ 9.8	$ 12.8	$ 159.6	$ 98.0	$ 11.2	$ 14.4	$ 168.2

* TOTAL PROGRAM INCLUDES 5 SPACECRAFT, 2 YR. OPS. FOR 4 UNITS, R&D, & 2 SETS OF MODULES FOR SHUTTLE.

Fig. 6-34 2-Year SEO Cost Comparison (1970 $ Millions)

Subsystem	RECOSTED BASELINE VS LCE				RECOSTED BASELINE VS SHUTTLE			
	RDT&E	Unit	Ops.	Total	RDT&E	Unit	Ops.	Total*
Experiments	13%	27%	9%	16%	20%	37%	16%	18%
Struct. & Mech.	39	54	11	44	43	54	22	47
Electrical	16	10	15	12	28	19	20	16
Stabiliz. & Control	35	20	15	28	44	24	30	26
Att. Control	5	(-17)	17	(-8)	17	-	17	1
CDPI	16	-	11	10	30	15	22	18
Environ. Control	9	-	-	14	27	-	-	19
Total **	19%	15%	10%	17%	29%	25%	20%	21%

* Shuttle total program includes two sets of non-redundant modules.

** Including Adpater & Non-Allocated Costs.

Fig. 6-35 % Cost Savings by Subsystem for SEO

COST	CONFIG.	LABOR ENG.	LABOR MFG.	LABOR P.A.	MATERIAL MATERIAL	MATERIAL EQUIP./PARTS	TOTAL COST
RATE	BASELINE	$68530	$14102	$11435	$1272	$25486	$120825
	LCE	60861	7682	7578	1161	20707	97989
	SHUTTLE	55371	6043	5798	1039	17449	85700
% SAVINGS	LCE	11%	46%	34%	9%	19%	19%
	SHUTTLE	19%	57%	49%	18%	32%	29%
UNIT *	BASELINE	$3367	$2637	$2491	$292	$4365	$13152
	LCE	2964	1452	2221	273	4259	11169
	SHUTTLE	2704	1367	1556	262	3941	9830
% SAVINGS	LCE	12%	45%	11%	7%	2%	15%
	SHUTTLE	20%	48%	38%	10%	10%	25%
OPERATIONS	BASELINE	$11749	—	—	—	$4304	$16053
	LCE	11027	—	—	—	3350	14377
	SHUTTLE	9443	—	—	—	3325	12768
% SAVINGS	LCE	6%	—	—	—	22%	10%
	SHUTTLE	20%	—	—	—	23%	20%
TOTAL OF ABOVE *	BASELINE	$83646	$16739	$13926	$1564	$34155	$150030
	LCE	74852	9134	9799	1434	28316	123535
	SHUTTLE	67518	7410	7354	1301	24715	108298
% SAVINGS	LCE	11%	45%	30%	8%	17%	18%
	SHUTTLE	19%	56%	47%	17%	28%	28%

* DOES NOT INCLUDE 4 ADDITIONAL SPACECRAFT AND NON-REDUNDANT

Fig. 6-36 SEO Functional Cost Comparison & % Savings (1970 $ Thousands)

largest percentage savings occur in manufacturing and product assurance. This is due to considerably simplified manufacturing of the payload and reduced testing of the ruggedized components (discussed above for the subsystem percentage savings comparison). The unit cost product assurance savings are not as large because, similarly to the OAO-B, the modular construction of the low-cost SEO requires testing at the module level. The equipment/parts and material cost savings are not as significant as in the OAO-B case, because SEO is a considerably simpler spacecraft and was not subjected to as severe a redesign as was the OAO-B. There are fewer part substitutions and redundant systems elimination.

The engineering cost reduction potential is not as large as for the OAO-B, because of: (1) lower complexity of the SEO payload; (2) the inclusion of $12M in engineering cost for the AVCS development is constant in all SEO cases, (3) the increased operations cost of two year mission support as opposed to one year in the OAO-B.

Additional savings analyses and comparisons with the other payloads studied are presented in Section 6.5.

6.4.3 SRS Cost Estimates

The SRS payload is representative of the small and fairly simple scientific satellites. The SRS baseline costs were extracted from the accounting records and subsequently adjusted to reflect an inclusive cost aggregation and elimination of learning effects.

The SRS program consists of development, four flight spacecraft and two years of operations at 6 mos. per spacecraft.

The SRS represents an austere spacecraft program approach. The P-11, from which SRS was derived, was developed with minimum development and qualification hardware, no structural test articles, minimum program controls, planning, support effort and minimum documentation.

6.4.3.1 Shuttle-Launched SRS. The low-cost SRS cost estimates for the Shuttle-launched case are shown in Fig. 6-37. The costs are tabulated by subsystem for RDT&E, 4 flight units, and operations for 4 units resulting in $14M total program cost. The costs shown for the experiment subsystem are the actual baseline costs converted into 1970 dollars. The operations costs are based on GFE checkout, GFE mission and launch operations.

The functional cost breakdown for the SRS payload is tabulated in Fig. 6-38. The development cost is the same as in previous figure, but only single unit hardware and operations costs are shown.

6.4.3.2 Expendable-Launched SRS. The LCE SRS program cost is based on the same hardware quantities and operating assumptions as the Shuttle-launched SRS. The LCE SRS costs by subsystem are presented in Fig. 6-39 showing $10.9M RDT&E cost estimate, $6.9M hardware cost estimate, and $0.8M operations cost for the four unit program.

The Payload Summary in Fig. 6-40 is the functional cost breakdown for the expendable-launched SRS. As in the Shuttle-launched case, the recurring production cost shown is for a single unit and the operations costs represent 6 mos. support for one payload.

6.4.3.3 Baseline SRS. Fig. 6-41 presents the revised baseline SRS program costs by subsystem. The total cost for a four flight program is $21.7M, composed of $12.7M in RDT&E, $8.2M in hardware, and $0.8M in operations.

The baseline costs were not recosted to permit the derivation of functional cost breakdown consistent with the low-cost payload cost classification. Therefore comparisons are only possible at the subsystem level.

6.4.3.4 SRS Cost Comparison. The estimated SRS costs are summarized in Fig. 6-42. The figures shown are for development cost, average unit cost, unit operations cost, and total program cost by individual subsystem.

SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS

(1970 $ THOUSANDS)

PAYLOAD: LOW-COST SRS LAUNCH VEHICLE: SPACE SHUTTLE FLIGHT DURATION: 6 MOS.

SUBSYSTEM	PROGRAM COSTS			
	RDT&E COST	4 UNIT COST	2 YEAR OPERATIONS	TOTAL COST
LAUNCH VEHICLE ADAPTER	—	—	—	—
EXPERIMENTS (GFE COSTS SHOWN)	360	320	440	1120
STRUCTURES & MECHANISMS	1086	436	44	1566
ELECTRICAL & PYROTECHNICS	2150	1640	44	3834
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL	1273	784	44	2101
PROPULSION & ATTITUDE CONTROL	808	320	68	1196
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)	1716	1876	132	3724
ENVIRONMENTAL CONTROL	297	192	—	489
NON-ALLOCATED COSTS	11	4	8	23
PAYLOAD [illegible]	$ 7701	$ 5572	$ 780	$ 14053

Fig. 6-37

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DETAILED COST ALLOCATION TO EACH SUBSYSTEM

(1970 $ THOUSANDS)

SUBSYSTEM SUMMARY PAYLOAD TYPE SRS- SPACE SHUTTLE

	COST CATEGORY	LABOR ENG.	LABOR MFG.	LABOR PA	MATERIEL MATERIAL	MATERIEL EQUIP/PARTS	SUB-TOTALS	TOTALS
RDT&E	PROGRAM MANAGEMENT	631	57	24	17	17	746	
	DESIGN/ENGINEERING	3787	—	—	—	—	3787	
	DEVELOPMENT & QUAL. TEST HARDWARE	189	127	—	52	58	426	
	~~FLIGHT TEST~~ EXPERIMENTS	—	—	—	—	360*	360*	$ 7701
	DEVELOPMENT TEST & QUALIFICATION	1336	36	110	53	—	1535	
	GRD. HDLG. EQUIP.	52	104	16	4	15	191	
	SUPPORT AND C/O EQ.	87	81	51	14	25	258	
	TOOLING	17	69	17	16	–	119	
	SPEC.TEST EQUIP.	113	86	16	11	53	279	
	SUBTOTAL	6212	560	234	167	528	7701	
RECURRING PRODUCTION	PROGRAM MGT.	14	14	5	5	94*	132*	
	MANUFACTURING	–	295	-	109	279	683	
	PROD. ASSUR.	–	–	88	—	-	88	$1393
	ACCEPT. TEST	240	32	41	18	–	331	
	TOOL. & GSE MAINT.	2	2	1	1	2	8	
	SUSTAIN. ENGR.	87	—	—	—	—	87	
	SPARES	—	—	—	10	54	64	
OPERATIONS	FLIGHT OPS. FLIGHT DATA REDUC.	—	—	—	—	110*	110*	
	TRANSPORTATION	–	—	—	—	2	2	$195
	CHECKOUT	–	–	—	—	—	—	
	LAUNCH OPS.	—	—	—	—	—	—	
	SUST.ENG & PROG MGT	83	—	—	—	–	83	
	TOTAL	6638	903	369	310	1069	9289	$9289

* INCLUDES GFE EXPERIMENT COSTS

Fig.6-38

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SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS
(1970 $ THOUSANDS)

PAYLOAD: LOW-COST SRS LAUNCH VEHICLE: LCE/ACE FLIGHT DURATION: 6 MOS

SUBSYSTEM	PROGRAM COSTS			
	RDT&E COST	4 UNIT COST	2 YEAR OPERATIONS	TOTAL COST
LAUNCH VEHICLE ADAPTER	588	524	8	1120
EXPERIMENTS (GFE COSTS SHOWN)	360	320	440	1120
STRUCTURES & MECHANISMS	1240	436	44	1720
ELECTRICAL & PYROTECHNICS	3008	1948	44	5000
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL	1887	892	44	2823
PROPULSION & ATTITUDE CONTROL	912	320	68	1300
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)	2527	2212	132	4871
ENVIRONMENTAL CONTROL	360	200	—	560
NON-ALLOCATED COSTS	11	16	8	35
PAYLOAD TOTAL	$10893	$6868	$788	$18549

Fig. 6-39

LMSC-A990556

DETAILED COST ALLOCATION TO EACH SUBSYSTEM
(1970 $ THOUSANDS)

SUBSYSTEM SUMMARY PAYLOAD TYPE SRS- LCE/ACE

	COST CATEGORY	LABOR ENG.	LABOR MFG.	LABOR PA	MATERIEL MATERIAL	MATERIEL EQUIP/PARTS	SUB-TOTALS	TOTALS
RDT&E	PROGRAM MANAGEMENT	760	136	55	46	75	1072	$10893
	DESIGN/ENGINEERING	4286	–	–	–	–	4286	
	DEVELOPMENT & QUAL. TEST HARDWARE	221	753	186	268	568	1996	
	~~[illegible]~~ EXPERIMENTS	–	–	–	–	360*	360*	
	DEVELOPMENT TEST & QUALIFICATION	1918	88	194	87	–	2287	
	GRD. HDLG. EQUIP.	56	111	17	5	18	207	
	SUPPORT AND C/O EQ.	87	81	51	14	25	258	
	TOOLING	23	84	21	20	–	148	
	SPEC.TEST EQUIP.	113	86	16	11	53	279	
	SUBTOTAL	7464	1339	540	451	1099	10893	
RECURRING PRODUCTION	PPOGRAM MGT.	15	19	7	7	96*	144*	$1717
	MANUFACTURING	–	420	–	144	340	904	
	PROD. ASSUR.	–	–	125	–	–	125	
	ACCEPT. TEST	269	32	46	18	–	365	
	TOOL. & GSE MAINT.	2	3	1	1	2	9	
	SUSTAIN. ENGR.	106	–	–	–	–	106	
	SPARES	–	–	–	10	54	64	
OPERATIONS	FLIGHT OPS. FLIGHT DATA REDUC.	–	–	–	–	110*	110*	$197
	TRANSPORTATION	–	–	–	–	2	2	
	CHECKOUT	–	–	–	–	–	–	
	LAUNCH OPS.	–	–	–	–	–	–	
	SUST.ENG & PROG MGT	85	–	–	–	–	85	
	TOTAL	7941	1813	719	631	1703	12807	12807

* INCLUDES GFE EXPERIMENT COSTS

Fig. 6-40

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SUBSYSTEM COST ESTIMATES FOR LOW-COST PAYLOADS
(1970 $ THOUSANDS)

PAYLOAD: SRS - REVISED BASELINE LAUNCH VEHICLE: PIGGY-BACK FLIGHT DURATION: 6 MOS

SUBSYSTEM	PROGRAM COSTS			
	RDT&E COST	4 UNIT COST	2 YEAR OPERATIONS	TOTAL COST
LAUNCH VEHICLE ADAPTER	500	520	8	1028
EXPERIMENTS (GFE COSTS SHOWN)	360	320	440	1120
STRUCTURES & MECHANISMS	1280	480	44	1804
ELECTRICAL & PYROTECHNICS	3950	2800	44	6794
GUIDANCE, NAVIGATION, STABILIZATION & CONTROL	2200	1200	44	3444
PROPULSION & ATTITUDE CONTROL	920	280	68	1268
TELEMETRY, TRACKING & COMMAND (INCL. INSTRUMENTATION, DATA PROCESSING, COMMUNICATIONS)	3000	2400	132	5532
ENVIRONMENTAL CONTROL	440	240	—	680
NON-ALLOCATED COSTS	10	—	8	18
PAYLOAD TOTAL	$12,660	$8,240	$788	$21688

Fig. 6-41

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SUBSYSTEM	BASELINE COSTS				SHUTTLE LAUNCHED SRS				EXPENDABLE LAUNCHED SRS			
	RDT&E	UNIT	OPS.	TOTAL PROGR.*	RDT&E	UNIT	OPS.	TOTAL PROGR.*	RDT&E	UNIT	OPS.	TOTAL PROGR.*
ADAPTER	$ 0.50	$ 0.13	—	$ 1.03	—	—	—	—	$ 0.59	$ 0.13	—	$ 1.12
EXPERIMENTS	0.36	0.08	$ 0.11	1.12	$ 0.36	$ 0.08	$ 0.11	$ 1.12	0.36	0.08	$ 0.11	1.12
STRUCT. & MECH.	1.28	0.12	0.01	1.80	1.09	0.11	0.01	1.57	1.24	0.11	0.01	1.72
ELECTRICAL	3.95	0.70	0.01	6.79	2.15	0.41	0.01	3.83	3.01	0.49	0.01	5.00
SCS	2.20	0.30	0.01	3.45	1.27	0.19	0.01	2.10	1.89	0.22	0.01	2.82
ACS	0.92	0.07	0.02	1.27	0.81	0.08	0.02	1.20	0.91	0.08	0.02	1.30
TT&C	3.00	0.60	0.03	5.53	1.72	0.47	0.03	3.72	2.53	0.56	0.03	4.87
ECS	0.44	0.06	—	0.68	0.30	0.05	—	0.49	0.36	0.05	—	0.56
UNALLOCATED	0.01	—	0.01	0.02	0.01	—	0.01	0.02	0.01	—	0.01	0.04
TOTAL	$ 12.66	$ 2.06	$ 0.20	$ 21.69	$ 7.71	$ 1.39	$ 0.20	$ 14.05	$ 10.90	$ 1.72	$ 0.20	$ 18.55

*Total program includes RDT&E cost, 4 spacecraft, and operations costs for 4 units.

Fig. 6-42 SRS Cost Comparison (1970 $ Millions)

The comparison between these costs was made in terms of percentage savings at the subsystem level and in total as shown in Fig. 6-43. At the total program level, 35 percent savings are estimated for the Shuttle-launched SRS and 14 percent savings for the expendable-launched SRS costs as related to the baseline SRS costs.

At the subsystem level, largest savings appear in the Electrical subsystem. These are due to use of more solar cells from a specific lot and elimination of the honeycomb solar panel structure with the cells mounted directly on the spacecraft structure of the low-cost payloads.

In the Shuttle case, the redundancies in the electrical subsystem have also been eliminated (i.e., battery and power control unit).

The savings in the SCS subsystem are due to ruggedized electronics and in the Shuttle case the savings above LCE is attributable to the elimination of auxiliary flight control electronics package.

In the CDPI subsystem, redundancies have been eliminated on the Shuttle-launched SRS and lower grade, heavier, components have been utilized with the resultant savings.

The experiment subsystem has been treated as a constant with no payload effects or cost changes. The operating costs are virtually the same with the exception of the elimination of the adapter in the Shuttle-launched SRS.

The SRS is incorporated in the further discussion of savings and payload effects analyses in Section 6.5.

Subsystem	BASELINE VS LCE				BASELINE VS SHUTTLE			
	RDT&E	Unit	Ops.	Total	RDT&E	Unit	Ops.	Total
Struct. & Mech.	3%	99%	-	5%	15%	9%	-	13%
Electrical	24	30	-	26	46	41	-	44
Stabilization & Control	14	26	-	18	42	35	-	39
Attitude Control	1	(-14)	-	(-3)	12	(-14)	-	6
CDPI	16	8	-	12	43	22	-	33
Environ. Control	18	17	-	18	32	20	-	28
Total*	14%	17%	-	14%	39%	32%	1%	35%

*Includes experiment costs which are constant, adapter & unallocated costs

Fig. 6-43 % Cost Savings by Subsystem for SRS

6.5 COST ANALYSIS AND IMPACT

In the previous section the detailed results of the cost analysis were presented and discussed for each of the three payloads studied. In this section comparisons are made between the payloads in order to: (1) identify the cost drivers, (2) quantify the payload effects, and (3) quantify the technology change contributions.

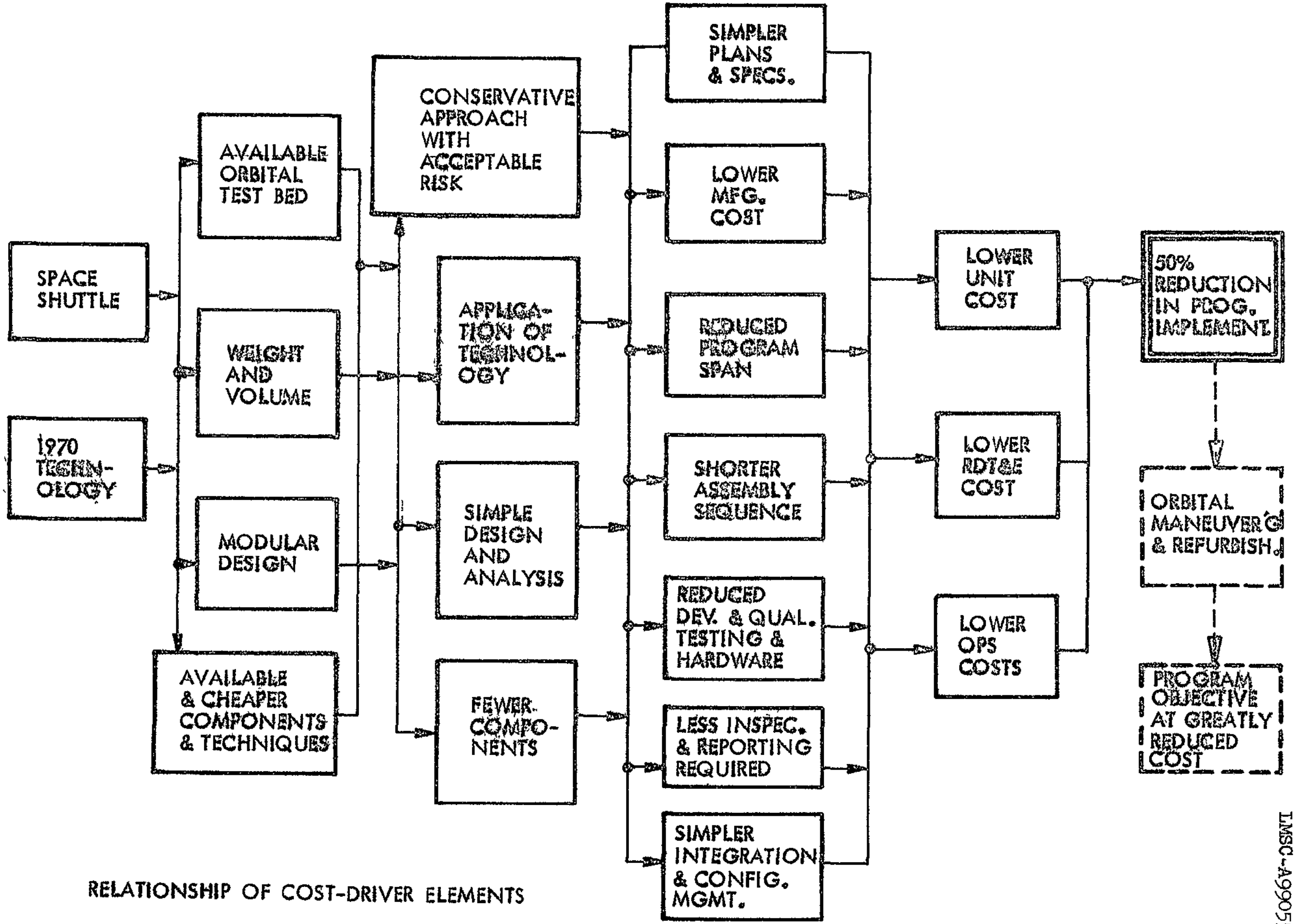
6.5.1 Cost Drivers

The potential of the Space Shuttle as a launch vehicle, orbiting test bed, and transportation system (enabling payload retrieval and refurbishment), combined with the application of 1970 technology; permitted a re-examination of conventional payload design, manufacturing, test and operations approaches.

In the design area, removal of stringent weight and volume constraints allowed employment of modular design, less sophisticated componentry, and ruggedization of the hardware. This results in simpler requirements for design and analysis and use of fewer components. The design impact is carried through into the manufacturing and test phases as shown in Fig. 6-44, resulting in lower costs for the total payload program.

6.5.1.1 Weight and Volume Constraints. The relaxed weight and volume impact on the OAO-B, SEO, and SRS, redesigned for low-cost, was considerable. Savings resulted in all major cost categories, as shown in Fig. 6-45 and in almost all subsystem costs as shown in Fig. 6-46.

The estimated savings in payload development costs ranged from 14% to 50%. The unit cost range was similar at 15% to 51%. The range of operations costs was from 0% to 51%. The resultant total program savings varied from 14% to 50% for the three payloads.



RELATIONSHIP OF COST-DRIVER ELEMENTS

Fig. 6-44

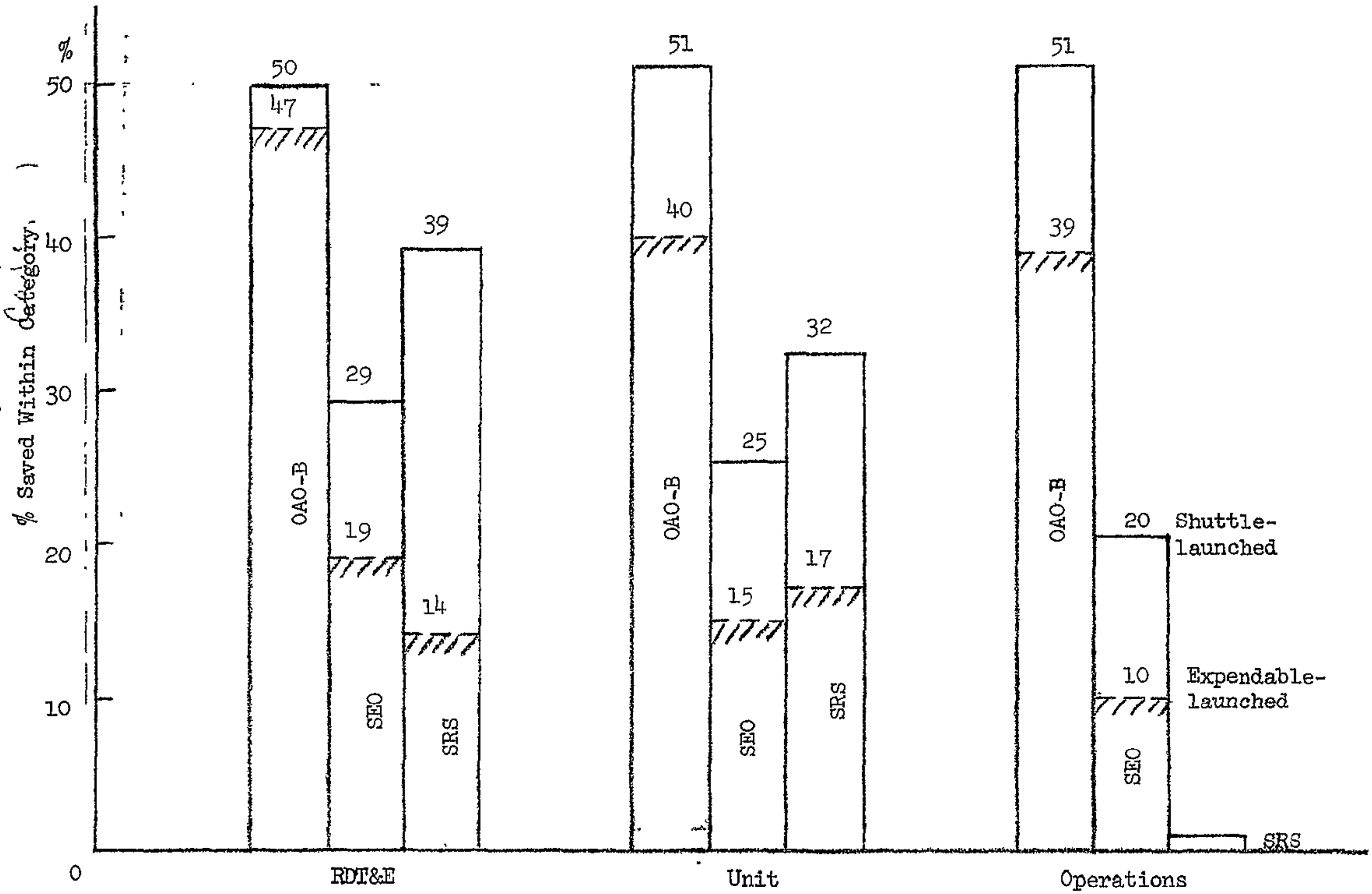


Fig. 6-45 Comparison of Savings within each Cost Category

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SUBSYSTEM	LCE – LAUNCHED PAYLOAD (% OF BASELINE COST)				SHUTTLE – LAUNCHED PAYLOAD (% OF BASELINE COST)			
	RDT&E	UNIT	OPS	TOTAL	RDT&E	UNIT	OPS.	TOTAL
EXPERIMENTS								
OAO-B	35%	14%	57%	33%	37%	18%	62%	37%
SEO	13	27	9	16	20	37	16	18
SRS	—	—	—	—	—	—	—	—
STRUCT & MECH.								
OAO-B	33	9	13	30	39	18	62	38
SEO	39	54	11	44	43	54	22	47
SRS	3	9	—	5	15	9	—	13
ELECTRICAL								
OAO-B	34	5	(-4)	28	35	22	1	32
SEO	16	10	15	12	28	19	20	16
SRS	24	30	—	26	46	41	—	44
STAB. & CONTROL								
OAO-B	60	59	56	59	63	67	67	64
SEO	35	20	15	28	44	24	30	26
SRS	14	26	—	18	42	35	—	39
ATT. CONTROL								
OAO-B	8	(-3)	14	6	12	6	14	11
SEO	5	(-17)	17	(-8)	17	—	17	1
SRS	1	(-14)	—	(-3)	12	(-14)	—	6
CDP & I								
OAO-B	44	47	24	43	46	60	31	47
SEO	16	—	11	10	30	15	22	18
SRS	16	8	—	12	43	22	—	33
ENV. CONTROL								
OAO-B	39	54	43	42	42	60	77	47
SEO	9	—	—	14	27	—	—	19
SRS	18	17	—	18	32	20	—	28
TOTAL PAYLOAD								
OAO-B	47%	40%	39%	45%	50%	51%	51%	50%
SEO	19	15	10	17	29	25	20	21
SRS	14	17	—	14	39	32	1	35

Fig. 6-46 Summary % Savings for Low-Cost Subsystems

At the subsystem level, even wider ranges were observed. The OAO-B stabilization and control and CDP&I subsystems, which were heavily impacted by the application of 1970 technology, showed considerable savings. The unit cost of the attitude control subsystem in all three payloads, as seen in Fig. 6-46, shows negative savings. This was caused by the subsystem requirement to support larger and heavier spacecraft with very few component changes, little redesign potential and no state-of-the-art advances. Further cost increases were caused by modularization to allow orbital repair and maintenance. Thus, the cost of the Attitude Control Subsystem can constitute a minor penalty in the low-cost payload approach. However, this penalty is minor compared to the off-setting savings afforded to the stabilization and control subsystem by oversizing of the Attitude Control Subsystem.

6.5.1.2 Simpler/Fewer Components. The use of simpler and fewer components constitutes another important cost savings contribution, because it impacts the unit cost, the developmental hardware cost and the basic design effort.

Fig. 6-47 is a specific example of the Communication, Data Processing and Instrumentation (CDPI) subsystem unit costs for the recosted baseline and the Shuttle-launched OAO-B. Thereon are shown: quantity of major components, estimated unit costs for each, and the breakdown between common, baseline peculiar, and low-cost peculiar equipment. The elimination of six components and about $3 million in baseline peculiar equipment costs, associated spares, test, assembly, integration, and program management; results in net subsystem savings of over $4 million per unit.

Similar, though less dramatic, savings due to fewer and simpler components were made in other subsystems permitting the aggregate 15% to 51% savings in unit cost.

6.5.1.3 Development Cost Drivers. The unit cost impacts the RDT&E cost directly in terms of the Development and Qualification Test Hardware and indirectly in the Design Engineering cost through component complexity and quantity.

EQUIPMENT	BASELINE		SHUTTLE-LAUNCHED	
	QUANTITY	UNIT COST	QUANTITY	UNIT COST
COMMON:				
COMMAND RECEIVER	4	$ 0.080 M	4	$ 0.080 M
NARROW-BAND TRANSMITTER	2	0.060	1	0.030
WIDE-BAND TRANSMITTER	2	0.300	1	0.150
DIPLEXER	2	0.040	2	0.040
HYBRID JUNCTIONS	2	0.010	2	0.010
ANTENNAS	4	0.060	4	0.060
SIGNAL CONDITIONING UNIT	1	0.130	1	0.070
INSTRUMENTATION SET	1	0.150	1	0.050
COMMON EQUIPMENT SUBTOT.	18	$ 0.830 M	16	$ 0.490 M
BASELINE PECULIAR:				
PRIMARY PROCESSOR & DATA STOR.	1	$ 2.370		
PROGRAMMER & SIG. CONTROLLER	1			
EXP. & S/C DATA HANDLING	2	0.515		
S/C SYSTEMS CONTROLLER	1	0.155		
AUX. COMMAND MEMORY	1	0.155		
TAPE RECORDER	1	0.100		
RADIO TRACKING BEACON	2	0.020		
B/L PECULIAR SUBTOTAL	9	$ 3.315 M	—	—
LOW-COST PECULIAR:				
COMPUTERS & INT. & TIMING UNITS			4	$ 0.720
DATA DISTRIBUTION UNIT			1	0.175
LOW-COST PECULIAR SUBTOTAL	—	—	5	$ 0.895 M
SUBSYSTEM:				
EQUIPMENT TOTAL	27	$ 4.145 M	21	$ 1.385 M
ASSY. TEST & INTEGRATION		1.425		0.833
SPARES		0.819		0.319
PROGRAM MANAGEMENT		0.532		0.211
TOTAL SUBSYSTEM	27	$ 6.921 M	21	$ 2.748 M

Fig. 6-47 OAO-B CDPI Equipment Comparison

The effect of unit cost reduction on SEO development and qualification test hardware is shown in Fig. 6-48. For the baseline test hardware cost, 2.3 equivalent SEO units were utilized. The shuttle-launched low-cost SEO development and qualification test hardware was based on only 1.5 equivalent units. However, even though the baseline hardware were used, 25% savings ($7.7 million) in development and qualification test hardware would be realized.

The Design Engineering cost relationship to unit cost is plotted in Fig. 6-49 for the baseline, LCE and shuttle cases. As shown in the plot, design engineering cost is related to unit cost by about a factor of three for payloads costing $10M or less, for higher cost payloads the curves flatten out and approach asymptotes.

The summary of payload cost drivers in terms of percentage cost contribution is illustrated for the OAO-B RDT&E and unit costs in Fig. 6-50. In RDT&E the sum of Design Engineering and Development and Qualification Test Hardware represents 65% to 68% of the cost. Support Equipment and Development and Qualification Test (separately) range from 10% to 15% and Program Management constitutes the remainder.

The unit cost is driven by purchased parts and manufacturing, which account for about half the payload cost, with sustaining engineering and acceptance test as the next to largest contributors. The spares and program management remained constant at 12% and 8% respectively in all the cases analyzed.

6.5.2 Space Shuttle and Expendable Launched Payload Savings

The identification of the major payload cost drivers discussed above provided the base for summarizing the savings represented by the low-cost payload designs. The estimated total program cost savings for the OAO, SEO, and SRS payloads are shown summarized in Fig. 6-51. More detailed breakdowns for the SRS are not available, since it was subjected to a less rigorous cost analysis.

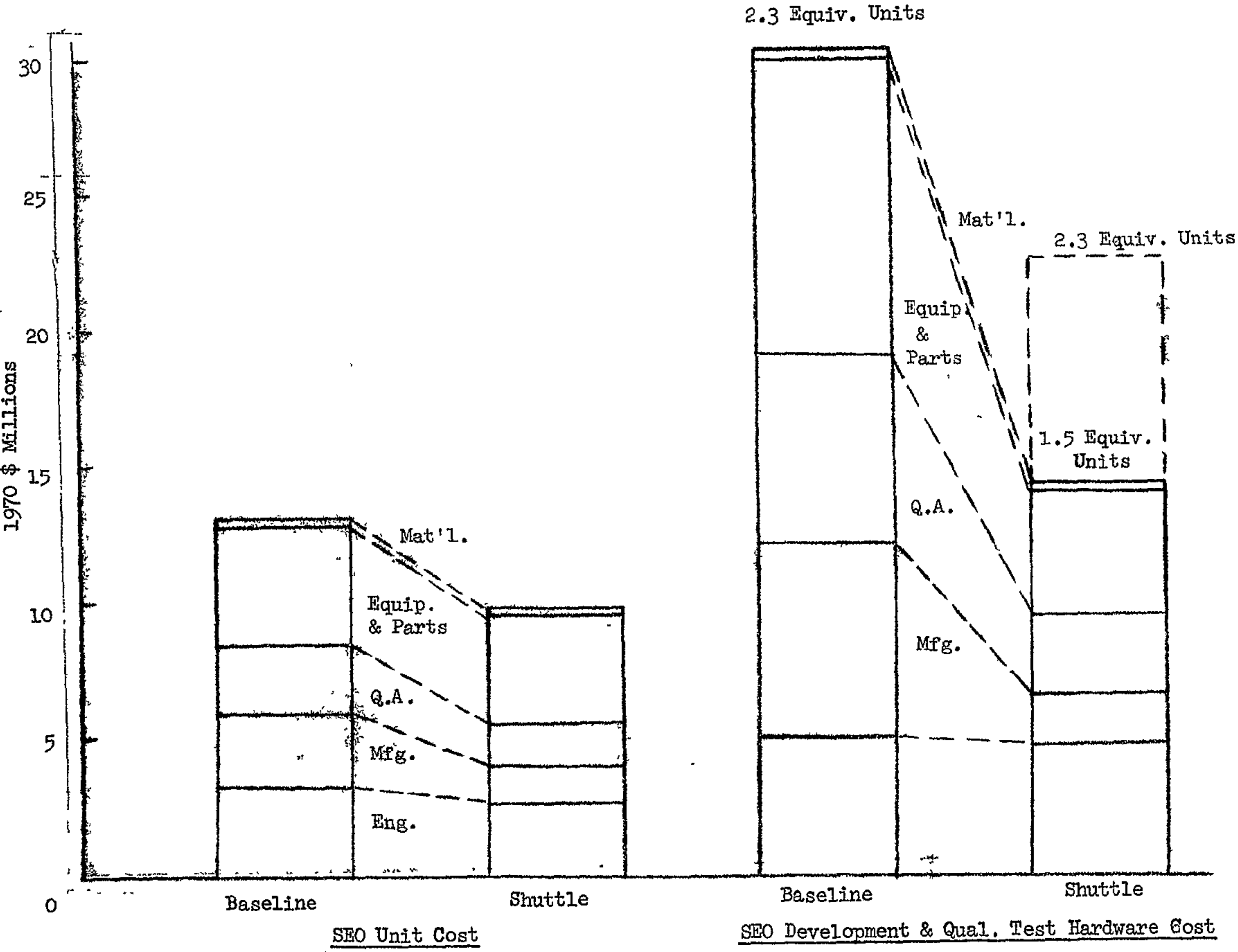

Fig. 6-48 Impact of Unit Cost on Development & Qual. Test Hardware
SEO Baseline vs Shuttle

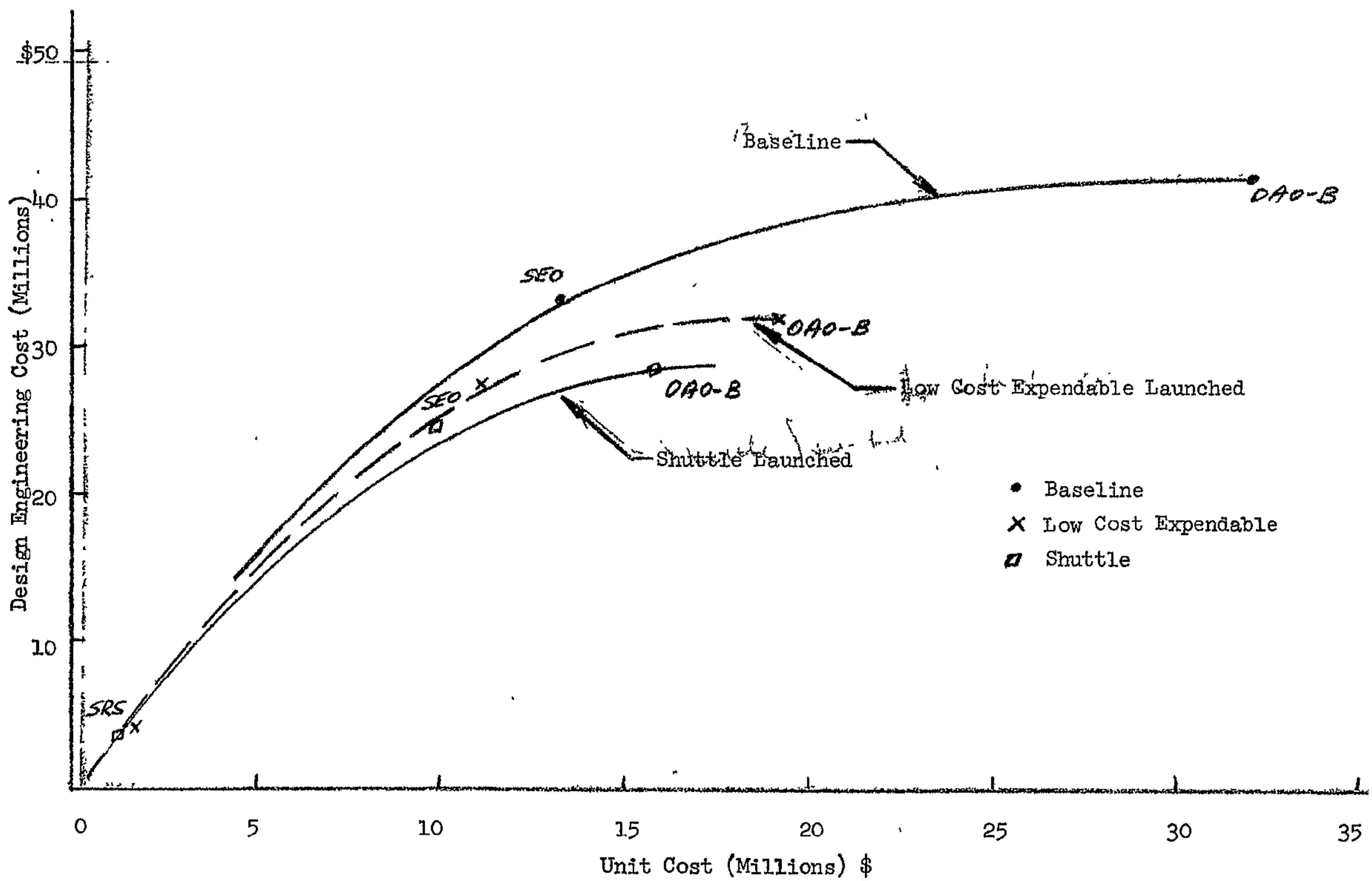

Fig. 6-49 Relationship of Unit Cost to Design Engineering

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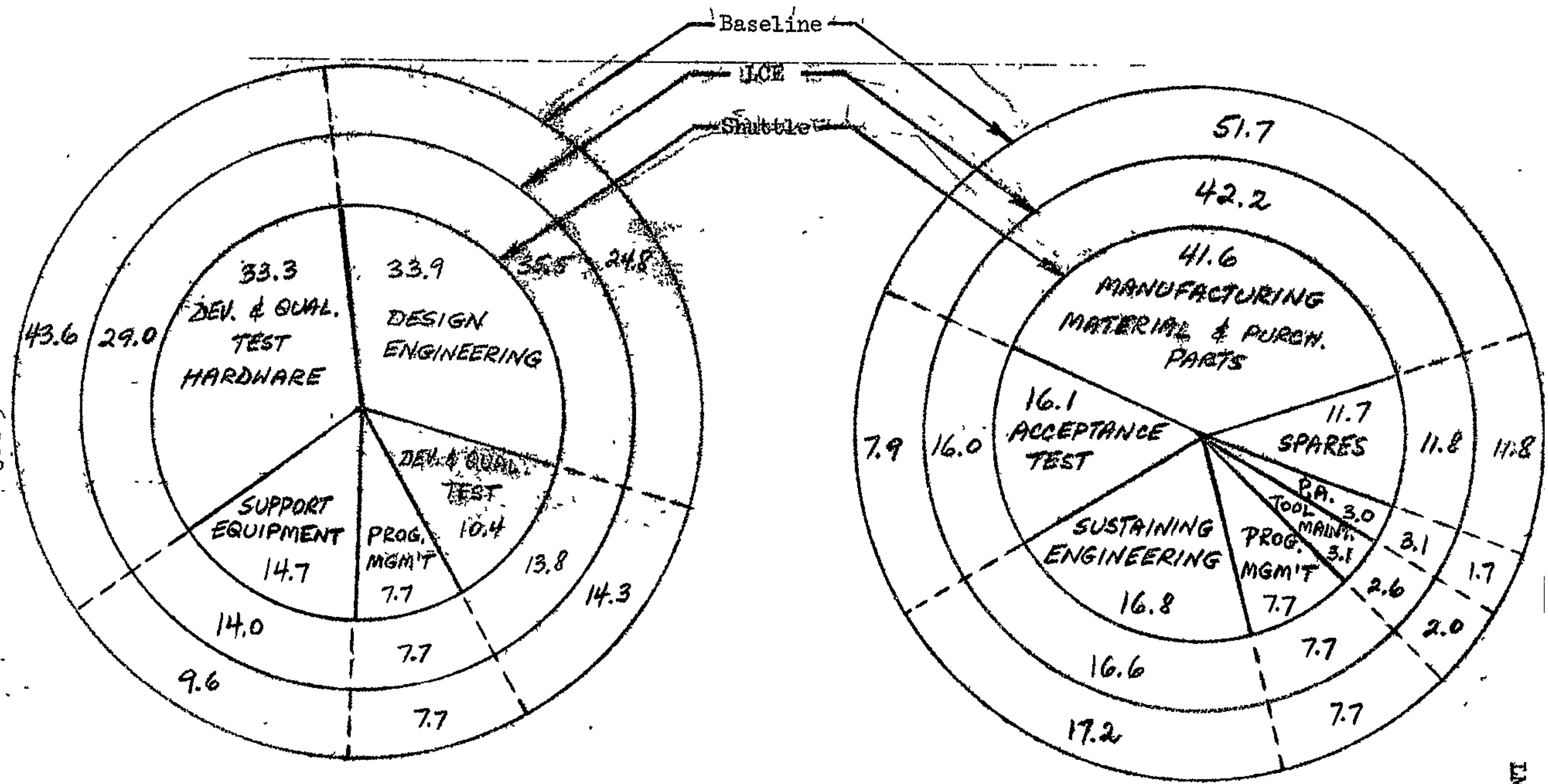

RDT&E

UNIT

Fig. 6-50 Payload Cost Contributors (%) - OAO-B

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COST CATEGORY	SHUTTLE-LAUNCHED LOW-COST PAYLOAD			L.C.E.-LAUNCHED LOW-COST PAYLOAD		
	OAO-B (1 Unit)	SEO (5 Units)	SRS (4 Units)	OAO-B (1 Unit)	SEO (5 Units)	SRS (4 Units)
NON-RECURRING COST						
o ENGINEERING	19%	21%		17%	22%	
o HARDWARE AND TEST	57%	35%	N.A.	62%	32%	N.A.
o SUPPORT EQUIPMENT	3%	8%		3%	12%	
SUBTOTAL	79%	64%	65%	82%	66%	56%
RECURRING UNIT COST						
o ENGINEERING	4%	5%	N.A.	3%	6%	N.A.
o MANUFACTURING & TEST	12%	25%		10%	23%	
SUBTOTAL	16%	30%	35%	13%	29%	44%
OPERATIONS						
o ENGINEERING	3%	2%	-	2%	3%	-
o MISSION OPS.	1%	-	-	2%	-	-
o PRE-LAUNCH OPS.	1%	4%	-	1%	2%	-
SUBTOTAL	5%	6%	-	5%	5%	-
TOTAL PROGRAM SAVINGS	100%	100%	100%	100%	100%	100%
SUMMARY						
o ENGINEERING	26%	28%		22%	31%	
o HARD.,MFG, & TEST	69%	60%	N.A.	72%	55%	N.A.
o OTHER	5%	12%		6%	14%	

NA = Not available.

Fig. 6-51 Program Cost Savings (As Percent of Baseline Costs)

As expected, the major cost reduction is in the hardware, manufacturing and test cost category; this is followed by engineering category. Support equipment and operations costs savings potential is rather negligible on a total program basis. The unit cost reduction is more sizeable in the SEO and SRS programs due to the fewer/simpler components in their design.

The estimated major cost reductions are highlighted in Fig. 6-52 for the OAO-B and SEO shuttle-launched cases. The sum of unit cost, development hardware and test cost, and the design engineering cost reductions account for 92% and 86% of total program cost savings for the OAO-B and SEO programs, respectively. The sources of the unit cost savings potential are listed in the lower half of Fig. 6-52 by major cost category and subsystem contribution.

A similar list of cost savings contributors to the non-recurring cost reduction is shown in Fig. 6-53.

The comparison of the low-cost expendable (LCE) and shuttle-launched payload savings by subsystem within RDT&E and unit cost is shown in Fig. 6-54 and Fig. 6-55. The OAO-B payload savings are dominated by the stabilization and control subsystem (S&C) cost reductions. The OAO stabilization and control subsystem cost reductions are detailed in Fig. 6-56, which shows the design changes made in that particular case and the resulting effects both in descriptive terms and dollars.

The SEO payload savings are primarily the result of the experiment cost reduction, although S&C and the CDP&I subsystems contribute approximately 40% of the RTD&E savings. In the case of the LCE/SEO unit cost, these were no measurable savings due to the CDP&I subsystem. The structures subsystem was the next most dominant savings contributor after the experiments.

The SRS payload derives the major portion of its savings from the electrical subsystem, which is very heavily impacted by the removal of extended solar paddles and body-mounting of the solar array panels integral with structure of the spacecraft. Section 6.4 discusses the characteristics of individual payloads and subsystems.

% Total Program Cost Reduction

	OAO-B	SEO	
Unit Cost	16%	30%	
Non-Recurring Hardware & Test Cost	57%	35%	
Non-Recurring Engineering Cost	19%	21%	
Above Account for	92%	86%	Reduction

Unit Cost Reduction

	OAO-B	SEO
Major Areas		
Manufacturing & Test:		
S&C	7%	5%
Comm. & Data Proc. & Instr.	3	4
Experiments	-	8
Structures	-	6
Total	10%	23%
Engineering:		
S&C	2%	1%
Comm. & Data Proc. & Instr.	1	1
Exp. & Electrical	1	2
Total	4%	4%
Other Areas	2%	3%
Total Unit Savings Contribution	16%	30%

Fig. 6-52 Major Cost Reductions (Baseline vs Shuttle)

MAJOR AREAS	% TOTAL PROGRAM COST REDUCTION OAO-B	SEO
Hardware & Test		
o S&C	34%	9%
o CDP&I	13%	8%
o Experiments	3%	7%
o Structures	1%	6%
o Electrical	4%	4%
o Other	2%	1%
TOTAL	57%	35%
Engineering		
o S&C	7%	4%
o CDP&I	4%	4%
o Experiments	3%	8%
o Other	5%	5%
TOTAL	19%	21%
ABOVE ACCOUNT FOR	76%	56%

OF TOTAL PROGRAM COST REDUCTION

Fig. 6-53 Non-Recurring Cost Reduction
(Compared with Baseline Cost)

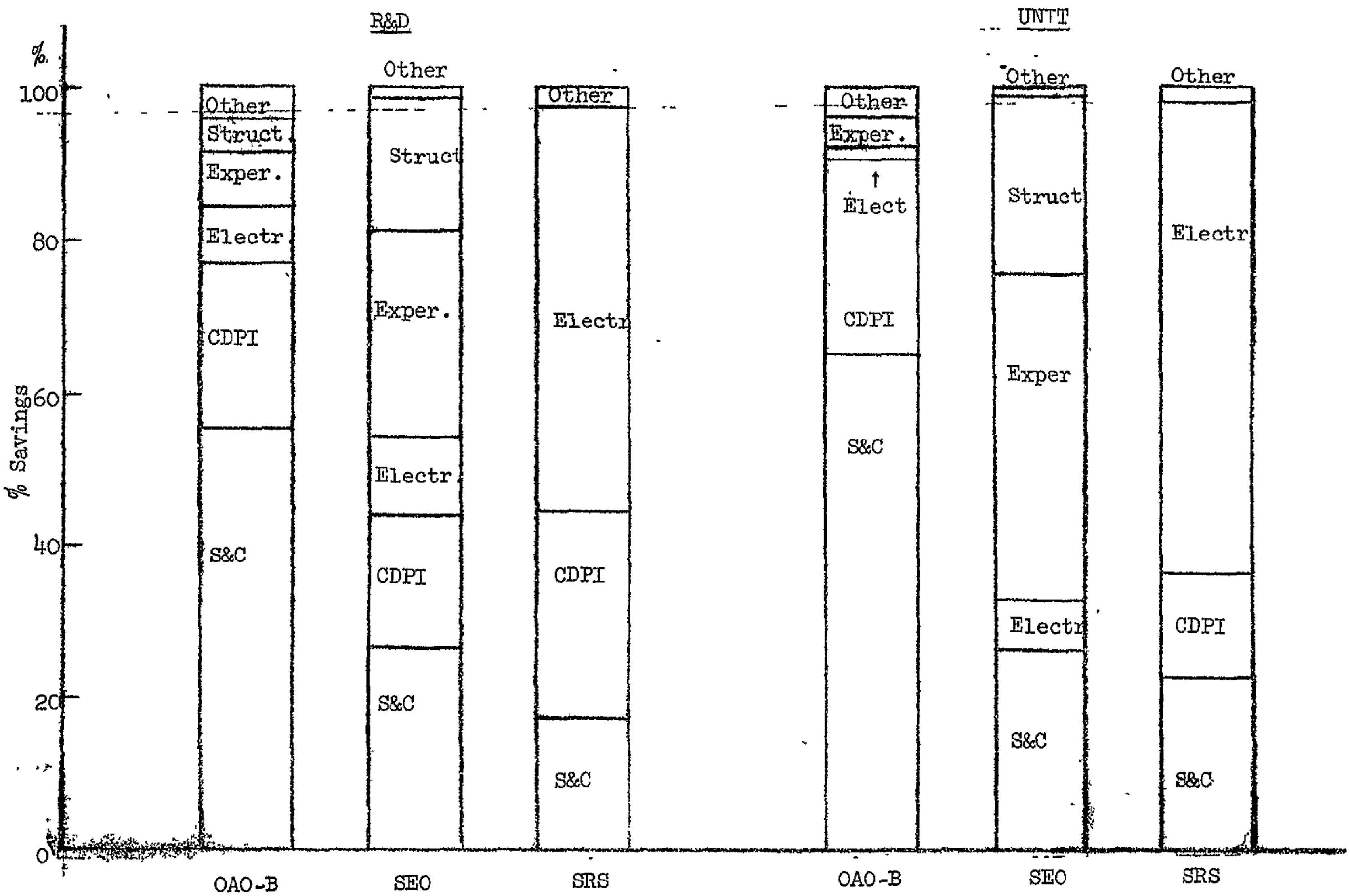

Fig. 6-54 % Savings Baseline vs LCE

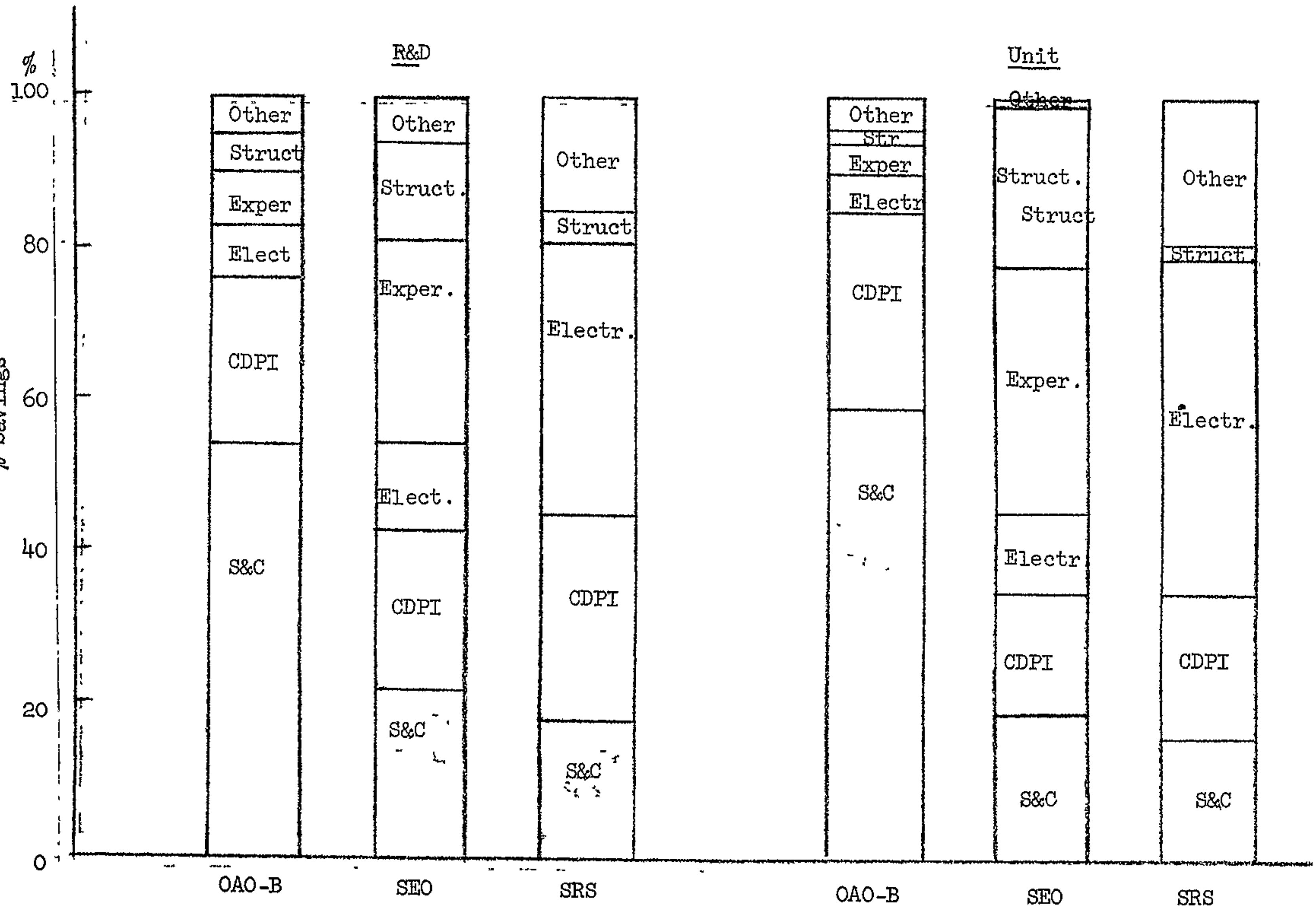

Fig. 6-55 % Savings Baseline vs Shuttle

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DESIGN CHANGES

- On-board computer added
- Substituted new IRU
- Substituted digital solar aspect sensors
- Removed rate gyro pkg.
- Cold gas used for slewing & unloading; eliminated coarse wheels and MUS
- Removed logic units & signal processors
- Decreased no. of Star Trackers
- Software vice hardware

DIRECTLY IMPACTS

- Subsystem design and analysis workload
- Procurement cost
- ACS design
- Manufacturing time and cost
- Development & qual. hardware
- Test planning and execution
- Sustaining engineering
- Development span

ANCILLARY EFFECTS

- Six expensive units in TT&C not required
- Influences on-board check-out approach
- Reduced ground/space operations & communications; hence mission ops costs
- Reduces checkout and validation span and costs for other subsystems

SAVINGS $:	RDT&E	UNIT	OPS	TOTAL
ENGINEERING	$ 7.8 M	$ 1.9 M	$ 1.1 M	$ 10.8 M
HARDWARE, MFG. & TEST	36.3 M	7.7 M	-	44.0 M
OTHER	1.2 M	-	1.3 M	2.5 M
TOTAL	$45.3 M	$ 9.6 M	$ 2.4 M	$ 57.3 M

Fig. 6-56 OAO S&C Subsystem Cost Impact (54% of Total Program Savings)(Example)

A closer look at the estimated potential savings within each of the primary RDT&E and unit cost categories is taken in Fig. 6-57 and Fig. 6-58. The OAO-B RDT&E cost savings are driven by the reduction in development and qualification test hardware of about $50M out of total $80M. The comparison of the LCE and shuttle-launched savings in OAO-B RDT&E reveals greater percentage reductions in all categories with the exception of Development Test and Qualification, which in the shuttle case has absorbed a $5M charge for the shuttle sortie test flight. This change in turn impacts the shuttle percentage cost savings when compared to the LCE case by about 3%, thus reducing the net shuttle savings to $5.4M or 3.2% from potential $10.4M or 6.2%.

The SEO RDT&E cost savings in Fig. 6-57 are derived primarily from the test hardware and development test costs, which account for half the savings in the LCE case. In the shuttle case, the test hardware contributes even a greater share of the savings due to the impact of lower unit cost (Fig. 6-58). The net RDT&E savings comparison of LCE vs. Shuttle SEO results in $12M or 10.2% advantage for the Shuttle.

The SRS baseline costs are not available at the detailed cost category level, thus the savings comparisons can only be made on basis of the RDT&E total cost resulting in $3.2M shuttle savings over the LCE. In the SRS case, the engineering cost represents about half the RDT&E cost. However, the shuttle savings are still driven by the development and qualification test hardware costs which are in turn driven by lower unit costs as shown in Fig. 6-58. The shuttle permits a 32% reduction in SRS unit cost vs. half as much in the SRS/LCE case.

The OAO-B unit cost savings appear in every unit cost category with the exception of acceptance test. This category is affected by the modular spacecraft construction, which requires acceptance testing at the module level. The LCE case, requiring more testing than the shuttle-launched payload, results in 21% negative savings. In total, the shuttle-launched OAO-B unit cost shows $3.3M or 10.4% additional savings over the low-cost expendable-launched OAO-B.

PAYLOAD	CONFIG.	ENGINEERING & PROG. MGM'T	DEV. & QUAL. TEST HARDWARE	DEV. TEST & QUALIFICATION	SUPPORT EQUIPMENT	TOTAL COST	SHUTTLE Δ SAVINGS
OAO-B	BASELINE	$ 54472	$ 73124	$ 24032	$ 16036	$ 167664	
	LCE	38613	23382	14838	12577	89410	
	SAVINGS $	$ 15859	$ 49742	$ 9194	$ 3459	$ 78254	
	%	29.1%	68.0%	38.3%	21.6%	46.7%	
	SHUTTLE	$ 34935	$ 19492	$ 17181	$ 12424	$ 84032	
	SAVINGS $	$ 19537	$ 53632	$ 6851	$ 3612	$ 83632	$ 5378
	%	35.9%	73.3%	28.5%	22.5%	49.9%	3.2%
SEO	BASELINE	$ 53599	$ 30330	$ 10361	$ 26535	$ 120825	
	LCE	46070	22551	7181	22187	97989	
	SAVINGS $	7529	$ 7779	$ 3180	$ 4348	$ 22836	
	%	14.0%	25.6%	30.7%	16.4%	18.9%	
	SHUTTLE	$ 42259	$ 14341	$ 7039	$ 22061	$ 85700	
	SAVINGS $	$ 11340	$ 15989	$ 3322	$ 4474	$ 35125	$ 12289
	%	21.2%	52.7%	32.1%	16.9%	29.1%	10.2%
SRS	BASELINE	N.A.	N.A.	N.A.	N.A.	$ 12660	
	LCE	$ 5718	$ 1996	$ 2287	$ 892	10893	
	SAVINGS $ %	N.A.	N.A.	N.A.	N.A.	$ 1767 14.0%	
	SHUTTLE	$ 4893	$ 426	$ 1535	$ 847	$ 7701	
	SAVINGS $ %	N.A.	N.A.	N.A.	N.A.	$ 4959 39.2%	$ 3192 25.2%

Fig. 6-57 Comparison of RDT&E Cost Savings (1970 $ Thousands)

NA = Not available.

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PAYLOAD	CONFIGURATION	SUST. ENG. & PROG. MGM'T	MFG. & PROD. ASSUR.	ACCEPT. TEST	SPARES & EQUIP. MAINT.	TOTAL COST	SHUTTLE Δ SAVINGS
OAO-B	BASELINE	$ 7947	$ 17084	$ 2530	$ 4404	$ 31965	
	LCE	4652	8680	3061	2742	19135	
	SAVINGS $	$ 3295	$ 8404	($ 531)	$ 1662	$ 12830	
	%	41.5%	49.2%	(21.0%)	37.7%	40.1%	
	SHUTTLE	$ 3875	$ 7049	$ 2554	$ 2336	$ 15814	
	SAVINGS $	$ 4072	$ 10035	($ 24)	$ 2068	$ 16151	$ 3321
	%	51.2%	58.7%	(0.9%)	47.0%	50.5%	10.4%
SEO	BASELINE	$ 2789	$ 6131	$ 2417	$ 1815	$ 13152	
	LCE	2382	4889	2380	1518	11169	
	SAVINGS $	$ 407	$ 1242	$ 37	$ 297	$ 1983	
	%	14.6%	20.3%	1.5%	16.4%	15.1%	
	SHUTTLE	$ 2235	$ 4544	$ 1696	$ 1355	$ 9830	
	SAVINGS $	$ 554	$ 1587	$ 721	$ 460	$ 3322	$ 1339
	%	19.9%	25.9%	29.8%	25.3%	25.3%	10.2%
SRS	BASELINE	N.A.	N.A.	N.A	N.A.	$ 2060	
	LCE	$ 250	$ 1029	$ 365	$ 73	1717	
	SAVINGS $	N.A.	N.A.	N.A.	N.A.	$ 343	
	%					16.7%	
	SHUTTLE	$ 219	$ 771	$ 331	$ 72	$ 1393	
	SAVINGS $	N.A.	N.A.	N.A.	N.A.	$ 667	$ 324
	%					32.4%	15.7%

Fig. 6-58 Comparison of Unit Cost Savings (1970 $ Thousands)

N.A. = Not available

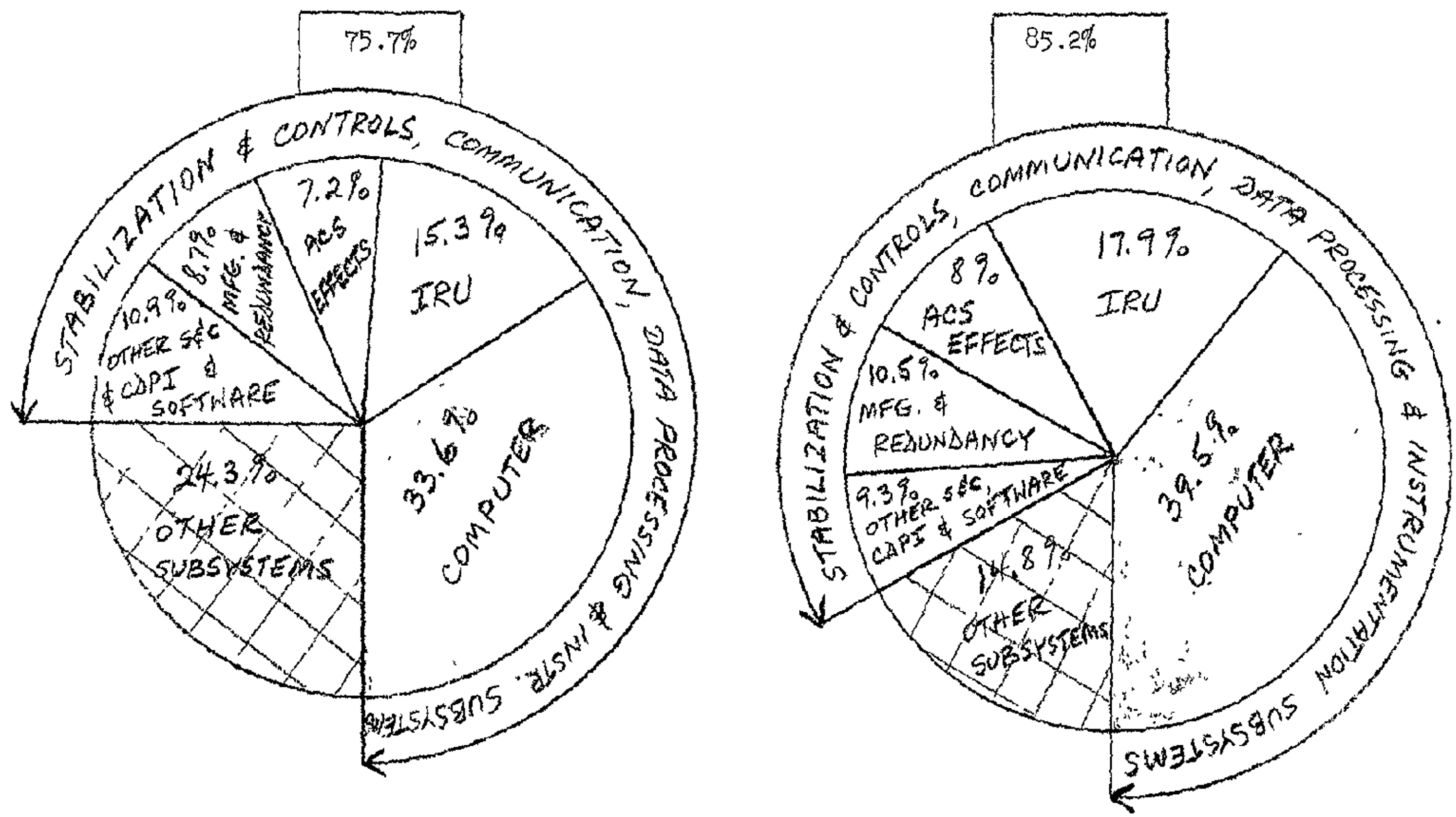
The simplified low-cost SEO modular spacecraft also requires acceptance testing at the module level; however, here the LCE costs are almost the same as baseline. In the shuttle case, savings were possible due to reduction in the quality assurance portion of acceptance testing. The cost reduction is about of the same order as for the OAO-B payload.

From Fig. 6-58 it is apparent that manufacturing, including purchased parts, material, and product assurance; is the largest unit cost savings contributer, especially if one keeps in mind that spares costs are derived directly from manufacturing cost category and program management is a percentage based on all the unit cost categories, including manufacturing,as its single largest component.

6.5.3 Effect of Technology

The previous data presented in this section include the cost savings due to incorporation of 1970 technology in the design of the low-cost payloads. This is particularly pertinent to the OAO-B payload costs, which were strongly impacted in the stabilization and control (S&C) and the communications, data processing, and instrumentation (CDPI) subsystems by the use of a digital computer in lieu of a number of complex electronic packages.

Figure 6-59 shows the contribution of these two subsystems, employing the computer, to the OAO-B shuttle-launched payload RDT&E and unit costs. In the RDT&E case, the S&C and CDP&I subsystems account for 75.7 percent of the estimated $83.6M RDT&E savings, or $63M. The 75.7 percent savings are composed, as shown in Fig. 6-59, of computer-utilization savings of 33.6 percent, the low-cost IRU substitution savings of 15.3 percent, increased attitude control system capability savings of 7.2 percent, low cost manufacturing and redundancy elimination savings of 8.7 percent, and software and other S&C and CDPI subsystem component savings of 10.9 percent. All the other subsystems combined contribute 24.3 percent ($20.3M) to the OAO-B total RDT&E cost savings.



RDT&E

Savings = 49.9% = $83.6 M

UNIT

Savings = 50.5% = $16.2M

(Shuttle-Launched)

Fig. 6-59 OAO-B Cost Savings Contributors with Computer

The S&C and CDPI subsystems, with the computer, impact the shuttle-launched OAO-B unit cost savings even more. As shown in Fig. 6-59, these two subsystems account for 85.2 percent of the unit savings or $13.8M. The computer represents savings of 39.5 percent ($6.4M). The other S&C and CDPI contributors add another 45.7 percent ($7.4M), and all the other subsystems combined account for $2.4M or 14.8 percent on the unit cost savings.

In summary, based on the estimated low-cost OAO-B shuttle-launched RDT&E and unit costs, the effects of 1970 technology represent 35 percent and the other payload effects represent 65 percent of the savings as shown in Fig. 6-60.

The removal of the effects of 1970 technology in the form of the OAO-B computer is shown in the unit and RDT&E cost comparison in Fig. 6-61. The savings in unit cost are reduced by 20 percent and the RDT&E cost savings decline by 16.8 percent in both the shuttle and expendable-launched cases.

Fig. 6-62 shows the comparison of SEO cost savings with the OAO-B, both including and excluding the effects of the computer or 1970 technology. Without the computer the savings are more comparable, averaging about 30% in RDT&E cost, about 27 percent in unit cost.

6.5.4 Refurbishment Cost Savings

Payload refurbishment constitutes the largest payload effect in terms of total payload program cost savings. The previous discussion was addressed primarily to payload RDT&E and unit cost savings, as well as program cost savings limited to one year or two-year program duration. The impact of refurbishment on payload program cost becomes apparent if programs of longer duration are considered (i.e. up to ten years).

For example, the analysis of OAO-B and SEO unit costs has shown the following unit cost ratios for refurbished vs. new spacecraft:

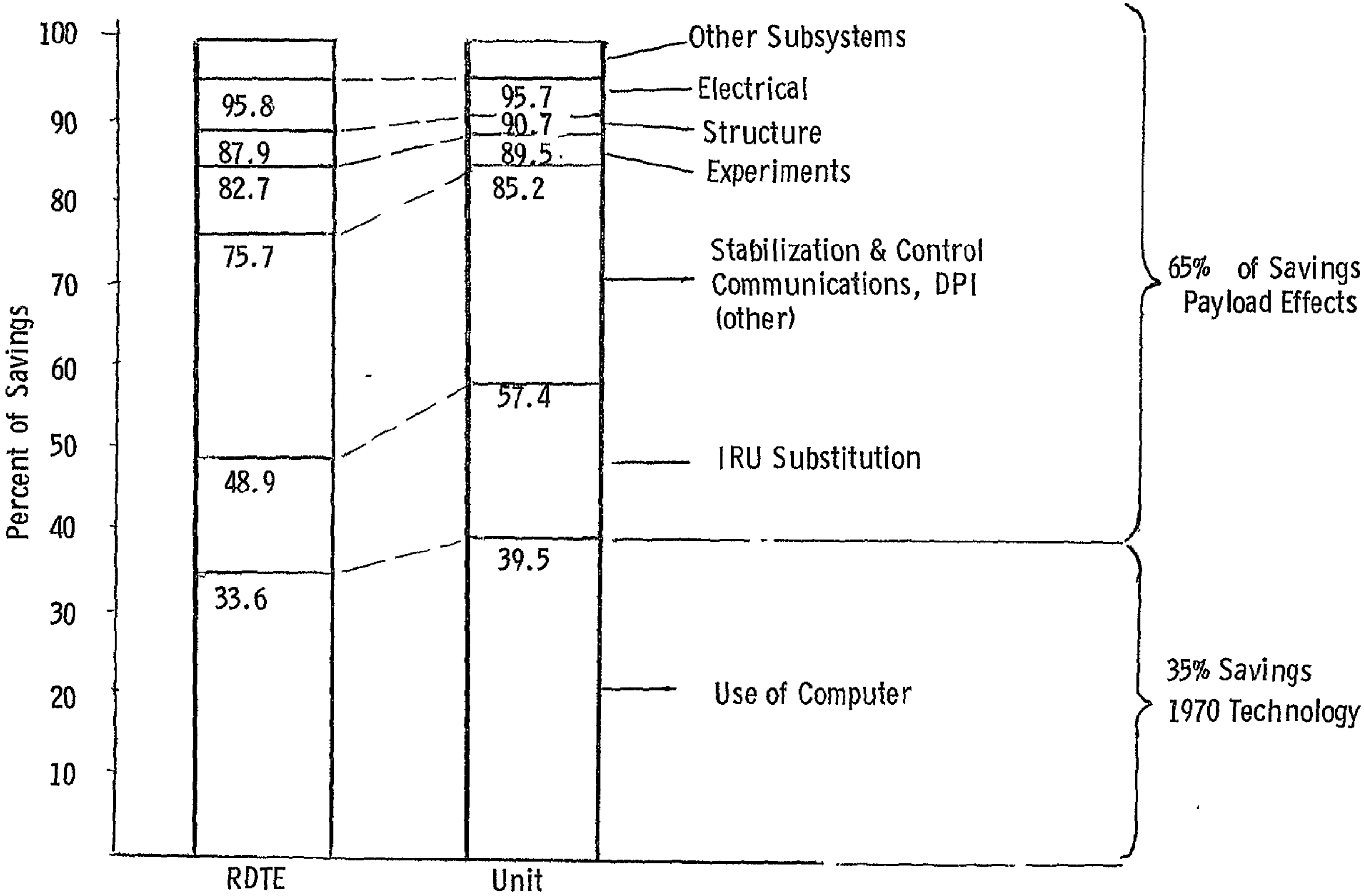


Fig. 6-60 Technology vs Payload Effects - OAO-B (Shuttle-Launched)

SUBSYSTEMS / COSTS	WITH COMPUTER				WITHOUT COMPUTER			
	LCE		SHUTTLE		LCE		SHUTTLE	
	RDT&E	UNIT	RDT&E	UNIT	RDT&E	UNIT	RDT&E	UNIT
STABILIZATION & CONTROL	$ 29.095	$ 5.902	$ 26.996	$ 4.669	$ 41.395	$ 9.052	$ 39.296	$ 7.819
COMM. DATA PROC. & INSTR.	21.904	3.671	20.903	2.748	37.704	6.921	36.703	5.998
SUBTOTAL	$ 50.999	$ 9.573	$ 47.899	$ 7.417	$ 79.099	$ 15.973	$ 75.999	$ 13.817
EXPERIMENTS	10.304	3.075	9.910	2.910	10.304	3.075	9.910	2.910
STRUCTURES & MECH.	7.362	1.050	6.688	0.947	7.362	1.050	6.688	0.947
ELECTRICAL	11.419	3.401	11.240	2.807	11.419	3.401	11.240	2.807
ATTITUDE CONTROL	4.496	1.100	4.296	1.014	4.496	1.100	4.296	1.014
ENVIRONMENTAL CONTR.	3.050	0.445	2.899	0.394	3.050	0.445	2.899	0.394
ADAPTER	0.580	0.141	—	—	0.580	0.141	—	—
NON-ALLOCATED COST	1.200	0.350	1.100	0.325	1.200	0.350	1.100	0.325
LC PAYLOAD TOTAL	$ 89.410	$ 19.135	$ 84.032	$ 15.814	$ 117.510	$ 25.535	$ 112.132	$ 22.214
RECOSTED BASELINE TOTAL	$ 167.664	$ 31.965	$ 167.664	$ 31.965	$ 167.664	$ 31.965	$ 167.664	$ 31.965
% COST SAVINGS	46.7%	40.1%	49.9%	50.5%	29.9%	20.1%	33.1%	30.5%
% SAVINGS DUE TO COMPUTER	16.8%	20.0%	16.8%	20.0%	—	—	—	—

Fig. 6-61 Computer Effect on Low-Cost OAO-B Costs (1970 $ Millions)

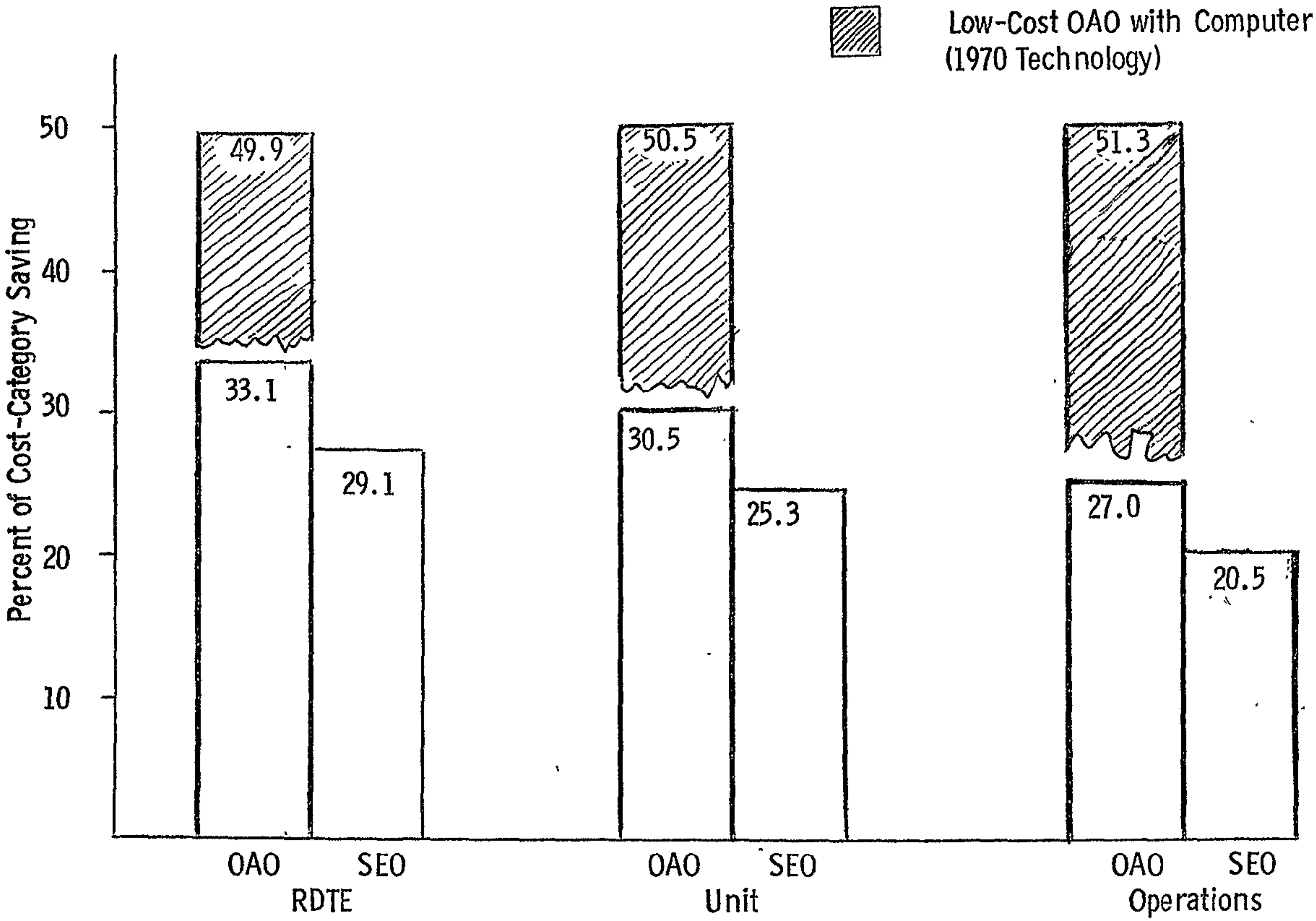


Fig. 6-62 Comparison of OAO with SEO Savings (Shuttle-Launched) (Example)

LMSC-A990556

$$\frac{\text{Refurbished OAO-B}}{\text{New OAO-B}} = \frac{\$\ 5.06M}{\$15.81M} = 0.325$$

$$\frac{\text{Refurbished SEO}}{\text{New SEO}} = \frac{\$3.81M}{\$9.83M} = 0.390$$

Combining the effects of the payload RDT&E and unit cost reductions and the use of the space shuttle in a longer duration payload program results in savings of 47 percent (SEO) to 62 percent (OAO) when compared to baseline program costs; - dollar comparisons are shown in Fig. 6-63 and Fig. 6-64. Figure 6-63 presents the cost comparison of six-year OAO-B program launched with SLV3C/Centaur in the baseline case, the TIII/L2 in the LCE case, and the Space Shuttle. The LCE case represents total savings of 23 percent and the shuttle 62 percent over the baseline program. The SEO ten-year program is shown in Fig. 6-64. Here, the LCE program results in negative total program savings of 5 percent due to the more expensive launch vehicle utilized. The shuttle-launched case, utilizing the space tug, provides 47 percent total program savings over the baseline case.

In the comparison of the shuttle-launched program to the low-cost expendable, total program costs for the shuttle-launched low-cost payloads are about 50 percent of the cost of an equivalent expendable-launched low-cost payload program. Detailed derivation of the refurbishment cost data is included in sub-section 8.4.

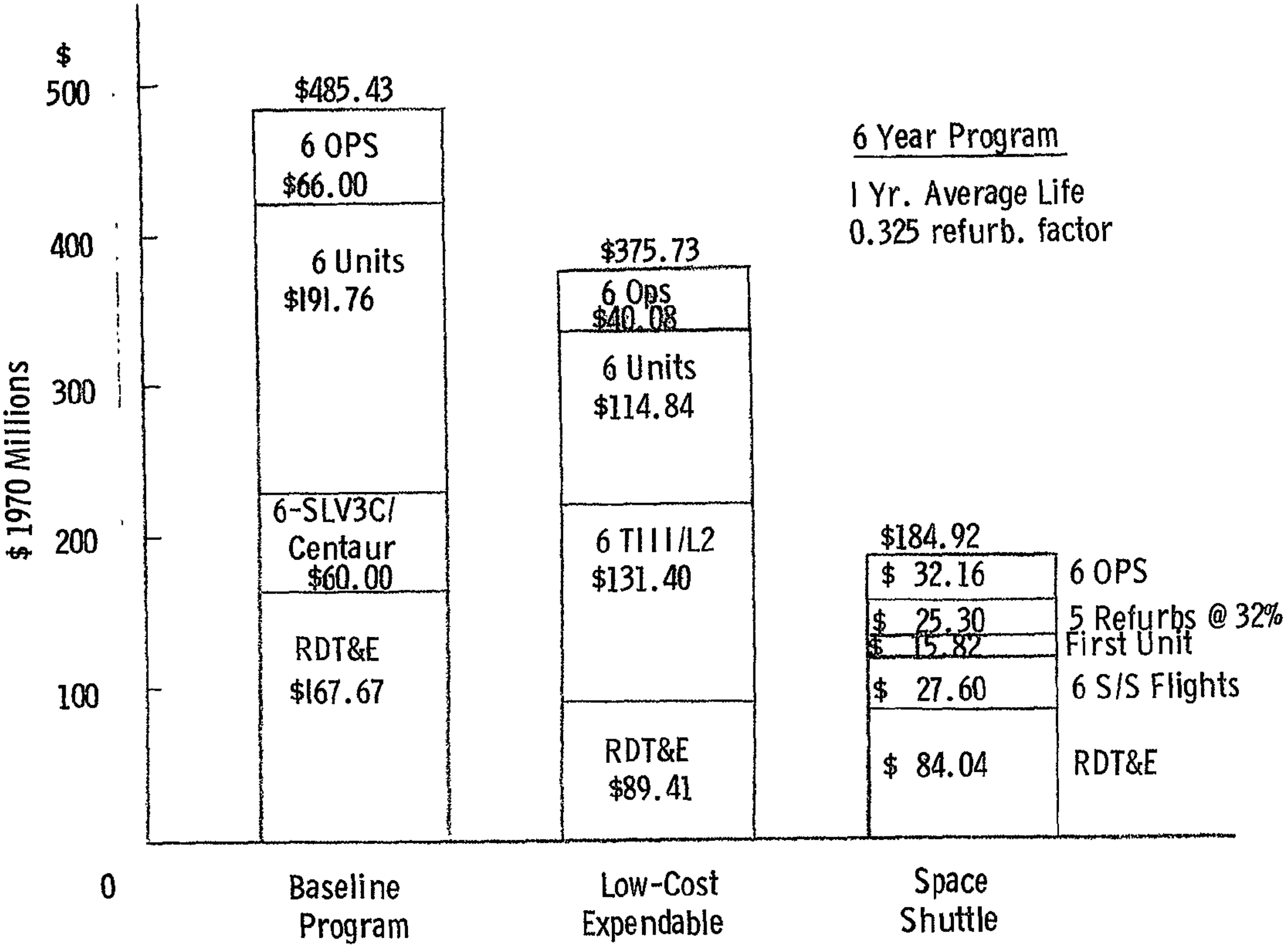


Fig. 6-63 OAO Total Program Cost Comparison

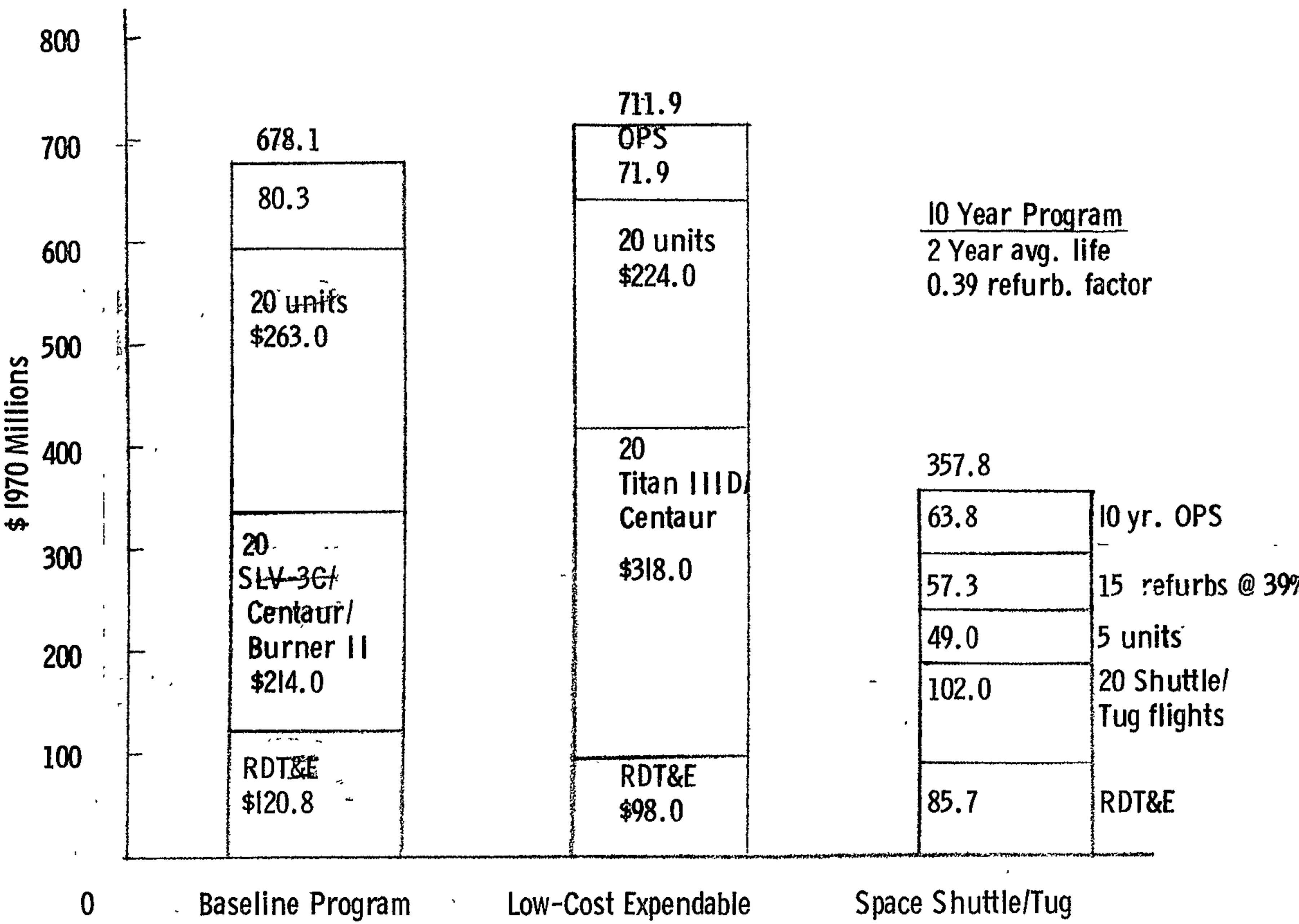


Fig. 6-64 SEO Total Program Cost Comparison

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Section 7
STANDARD SPACECRAFT AND SUBSYSTEMS

This section summarizes the work performed under Task 3 of the study, "Standard Spacecraft Design and Analysis". In view of the limited level of effort available for this task, the emphasis was placed on identification of economic and operational gross effects resulting from spacecraft and subsystem standardization rather than on optimizing these effects. The contention is that if standardization can be identified as a viable element of a low-cost payload design rationale, then the savings obtainable from a later optimized standardization must exceed the savings estimated herein. The previously-discussed low-cost payload design and economic analysis activity showed that the low-cost designs had very little impact on the shuttle design or on transportation system economics. Therefore, in a first approximation; minimization of the spacecraft cost is tantamount to minimization of program cost. All of the other program cost-reduction benefits, such as those accruing from refurbishment and reuse of payload hardware, are equally applicable to standard spacecraft hardware.

After discussing the various aspects by which standardization influences program costs, the two strongest effects, namely (1) development cost sharing and (2) matching of mission requirements with standard spacecraft hardware; are explored in a mission model capture analysis. "Reasonable" increments of subsystem performance used in the capture analysis are updated in a subsequent evaluation of standardized subsystems and spacecraft design approaches. The results of this analysis are then fed as inputs into a cursory assessment of the cost savings to be expected from standardization. Finally, implications of standardization for future spacecraft are discussed.

7.1 GENERAL ECONOMIC ASPECTS OF SPACECRAFT STANDARDIZATION

Standardization allows replacement of a large quantity and variety of program-peculiar developments with a limited quantity of standardized developments. If implemented at the proper systems level it will greatly use the design and integration of future spacecraft and will eliminate a large part of otherwise multiple development efforts.

The primary cost savings due to standardization derive from the sharing of development costs among a relatively large quantity of similar activities. These savings, however, are reduced by the need to develop versatile, multi-usage interfaces and to ensure that the hardware elements are functionally compatible in the many potential applications. Secondly, if only a limited number of options exist to cover a given program performance spectrum, a certain amount of overdesign must be taken into account. The mission model capture analysis indicates that standardization should be implemented in a way which allows a modular build-up of systems capabilities, rather than to implement a number of multipurpose spacecraft. There is, however, the strong probability that, given an inventory of modularized subsystems options, a multipurpose spacecraft could be assembled to take care of a local concentration of similar mission requirements. Therefore, three approaches were explored, using a set of standardized subsystem options:

(1) Design a single-purpose spacecraft as closely as possible to these required performance levels;

(2) Combine the requirements of a variety of missions into a composite that will be satisfied by a super-spacecraft as the only mode of standardization; and

(3) Use a super-spacecraft in conjunction with standard-based special purpose spacecraft.

These choices have several economic implications. As long as only the cost savings due to R&D sharing are considered, there is an incentive to minimize

the number of new spacecraft developments (i.e., the discrete R&D efforts beyond subsystems R&D required for spacecraft integration), and would favor the development of a few superspacecraft. A counteracting effect exists in excess recurring unit costs due to overdesign. Further studies are required to find the right balance. It appears that, if mission objectives could be coordinated such that the superspacecraft concept could be extended to accommodate several experiments at the same time; the advantages of R&D sharing, equipment sharing, and transportation and maintenance cost sharing could be combined to obtain a very significant cost savings effect.

7.2 PRINCIPLES OF SPACECRAFT STANDARDIZATION

The development of previous and present spacecraft is characterized by the repetitive procurement of rather similar subsystems. Large subsystems development costs are spread over only a small number of flight units purchased. It is therefore of considerable interest to low-cost payload design to explore the cost saving potential inherent in standardization of hardware. The question of equipment performance requirements-matching arises immediately; it has a direct bearing on the number of discrete subsystem design points necessary to cover a given performance requirements spectrum. While this may appear to involve straightforward arithmetic, efforts to standardize have historically had to stand the test of political/utilitarian considerations having to do with the freedom of choice and individuality of the user, and the potential stagnation of technology advancement in the very activity it was supposed to serve. It is, therefore, desirable to consider some of the criteria for and lessons learned in industrial standardization.

7.2.1 Criteria for Standardization

In order to produce useful standardization, the following criteria must be met:

- Consensus on Requirements

- Standardize only if a standard reflects a recurring requirement.
- Standardize such that unnecessary constraints are avoided

- Standardization by Evolution
  - Standardize fundamentals first
  - Standardize higher-level products only if they are in general use and have achieved a level of maturity

- Solution to Recurring Problems
  - Be sure that requirements and their particular solutions are indeed recurring often enough

- Profitability
  - Standards have actual value only when the money invested in them is recouped with interest

7.2.2 Historical Experience in Industrial Standardization vs. Aerospace Application

In general, industrial standardization is aimed at applications involving relatively high production numbers. It can, therefore, expect to save money both from the amortization of development costs as well as from amortization of production facilities and economies on labor and materials by means of mass production. Spacecraft procurement, by contrast, usually involves only a small number of vehicles of a kind. An appreciable degree of commonality exists at the part or component level; this is already recognized in the form of preferred and/or standard parts lists. A much greater share of total spacecraft program costs can be saved in the higher categories of subsystems and integrated spacecraft RDTE costs. This indicates strongly that spacecraft design must be changed from a unitized to a modular approach. Low-cost design principles for spacecraft, as developed elsewhere in this report, specify that spacecraft will be designed for on-orbit maintenance and repair. Again, modular design with easy access is necessary.

With a limited interchangeability of spares among several programs, a degree of quantity production and related savings may be realized. This effect will, however, be weakened if spares are procured on demand rather than on a forecast basis. Any inventory of standard parts is potentially an economic liability unless the product has previously achieved acceptance as a mature solution to a recurring requirement which thereby precludes early obsolescence.

Basically, standardization reduces the variance of items stocked, and will reduce the variants of performance options unless standardization is implemented at sufficiently low systems level to allow modular performance matching. It seems, therefore, that the emphasis must be on the standardization of adaptable interfaces both in an electronic and physical sense to allow a "tinker toy" assembly approach to spacecraft using standard equipment modules. Easy access to and replaceability of modules is required also for repair and maintenance purposes.

A given requirements spectrum can be covered by use of an optimum quantity of discrete subsystem designs with "tailoring" supplied by addition or deletion of certain submodules. Although this approach will accommodate the majority of the different mission requirements, there will probably remain a small group of special mission requirements which are more adequately covered by special program-peculiar subsystems. Even these latter subsystems should incorporate standard interfaces so that they can be matched with other standard subsystems and facilitate overall spacecraft integration.

7.3 MISSION MODEL CAPTURE ANALYSIS

The potential benefit of a standard spacecraft derives primarily from reduction of the development costs associated with any given program by using available, developed hardware. This available hardware will in general have excess capability for the particular mission and in some cases may have higher unit costs. Higher unit costs will offset some of the reduction in development cost to an extent which depends on the number of spacecraft required for the program.

In choosing a candidate standard spacecraft, or a subsystem for such a spacecraft, one therefore seeks to provide enough capability to capture a substantial proportion of the more demanding missions without becoming uneconomic for the simpler ones. For these reasons the following analyses, while primarily addressed to technological capture, were at all times performed with economic considerations in mind.

7.3.1 Basic Approach

The approach adopted was as follows:

a. Tabulate key performance requirements for each mission in the NASA Mission Model for unmanned payloads

b. Analyze populations of numbers of programs and quantity of flights versus required spacecraft subsystem performance capability

c. Postulate levels of subsystem performance capability which would capture a substantial proportion of the programs, based on economic as well as performance criteria

d. Postulate corresponding subsystems hardware options, reducing the variants to a minimum quantity

e. Postulate standard spacecraft(s) as a combination of standard subsystem options

f. Propose first-cut standard subsystem and standard spacecraft candidates as a starting point for design studies

7.3.2 Analysis of Subsystem Performance Requirements

Performance requirements for the key, cost-driving subsystems were taken from mission payload data supplied by Aerospace Corp.* which was a digest of the

* Aerospace Report No. ATR-71(7231)-7, "Payload Data for Payload and Interface Analysis Subtask"

NASA Mission Model data. In the cases for which no relevant data were contained in the reference, entries were made based on Lockheed background for similar missions.

The requirements for pointing reference accuracy (stabilization and control) are those which the spacecraft guidance system must meet or exceed to enable the experiment-sensor to acquire the target. After acquisition more refined pointing would be achieved using error signals generated by the sensor. Thus a stellar telescope might acquire a target star based on a pointing reference accurate to 10 $\widehat{sec}$ but might then hold the line of sight to star to better than one $\widehat{sec}$, with very low angular rates, based on error signals generated by the experiment equipment. The requirements listed for the wideband spacecraft to ground data link do not include the transponder function of the pure communication relay mission. In a satellite which is purely a communication relay, this repeater function is performed by the "experiment" without imposing any requirement for a wide band data link on the support spacecraft. There is, however, still a requirement for housekeeping telemetry on the narrow band (VHF) link.

Cumulative distribution curves are presented in Figs. 7-1, 7-2, and 7-3 of the number of programs captures (and the associated number of flights) versus subsystem performance for pointing reference accuracy, electrical power and experiment weight. Characteristically, these curves show that a fairly large proportion of the missions (65 percent or better) is captured by a subsystem of moderate capability, with rapidly decreasing marginal returns as subsystem capability is further enlarged. Groupings of wideband data link requirements are shown in Fig. 7-4; somewhat narrower bands for bit rate have been estimated than those from the aforementioned Aerospace report. In summary, (1) a pointing reference accuracy of 0.2 deg is sufficient for 36 out of 53 programs (68 percent) whereas an accuracy of 0.02 deg would capture 46 (87 percent); (2) an average electrical power capability of 1050 watts is sufficient for 43 programs (81 percent).

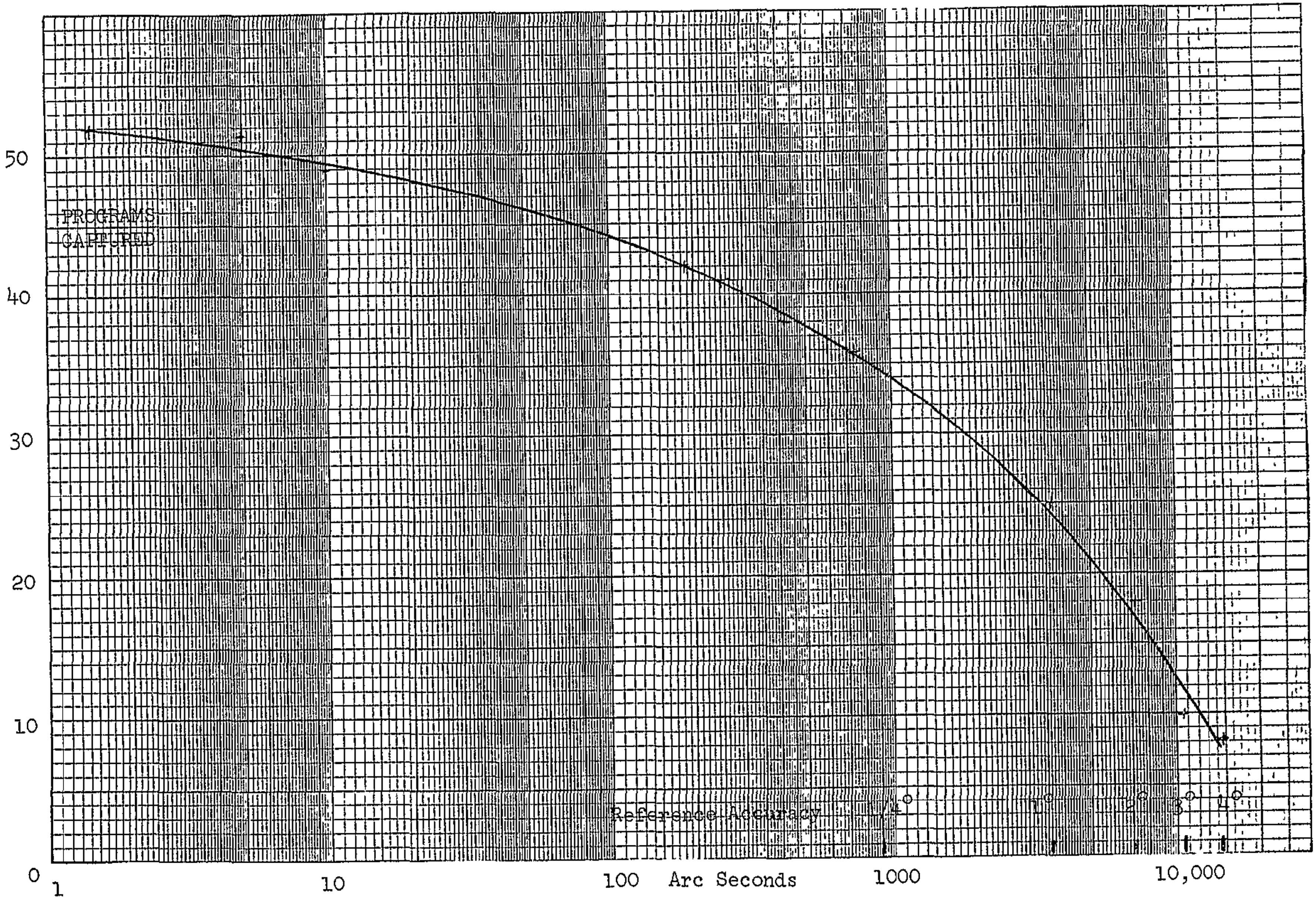


Fig. 7-1 Programs Captured vs Pointing Reference Accuracy

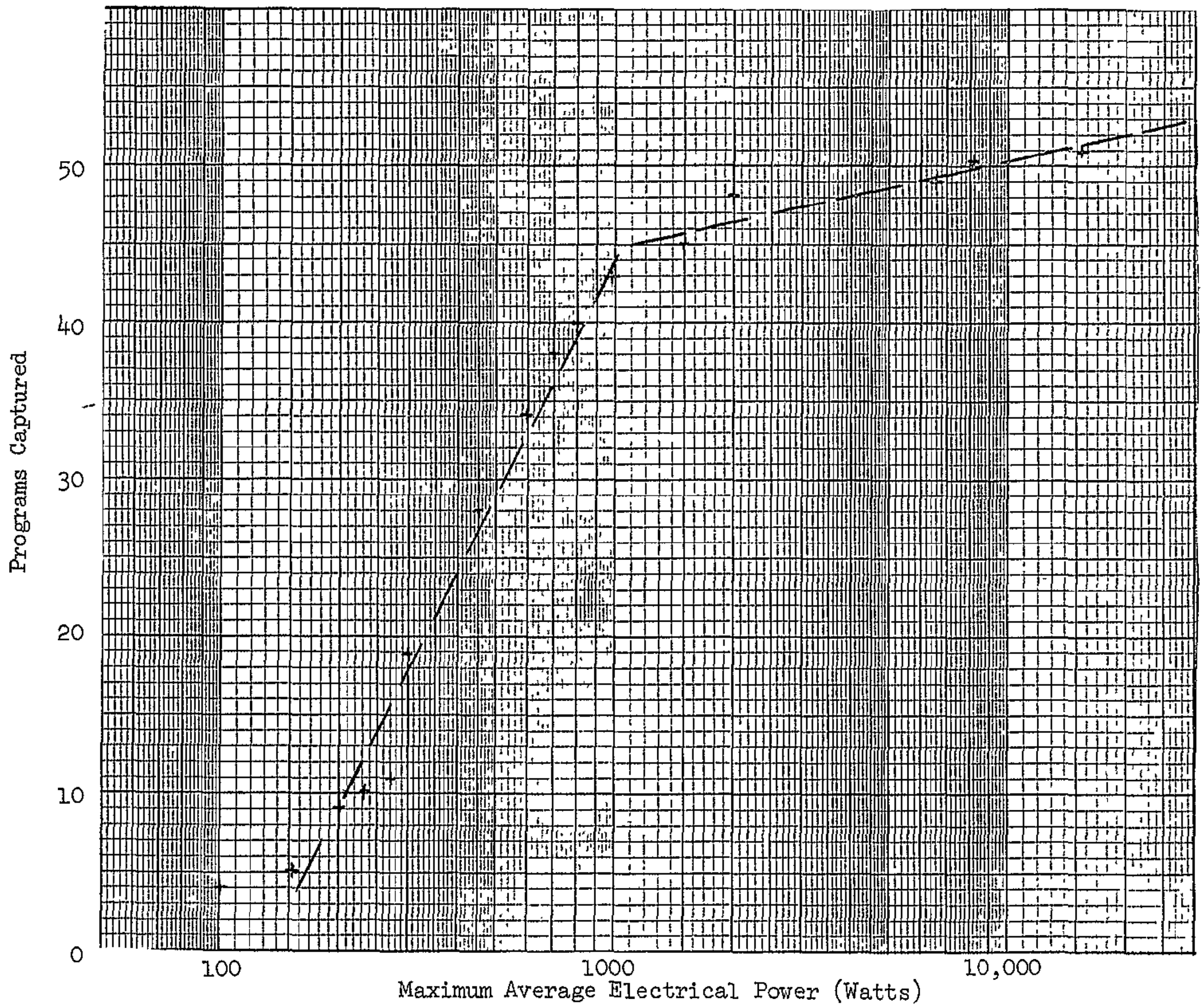


Fig. 7-2 Programs Captured vs Maximum Avg. Electrical Power

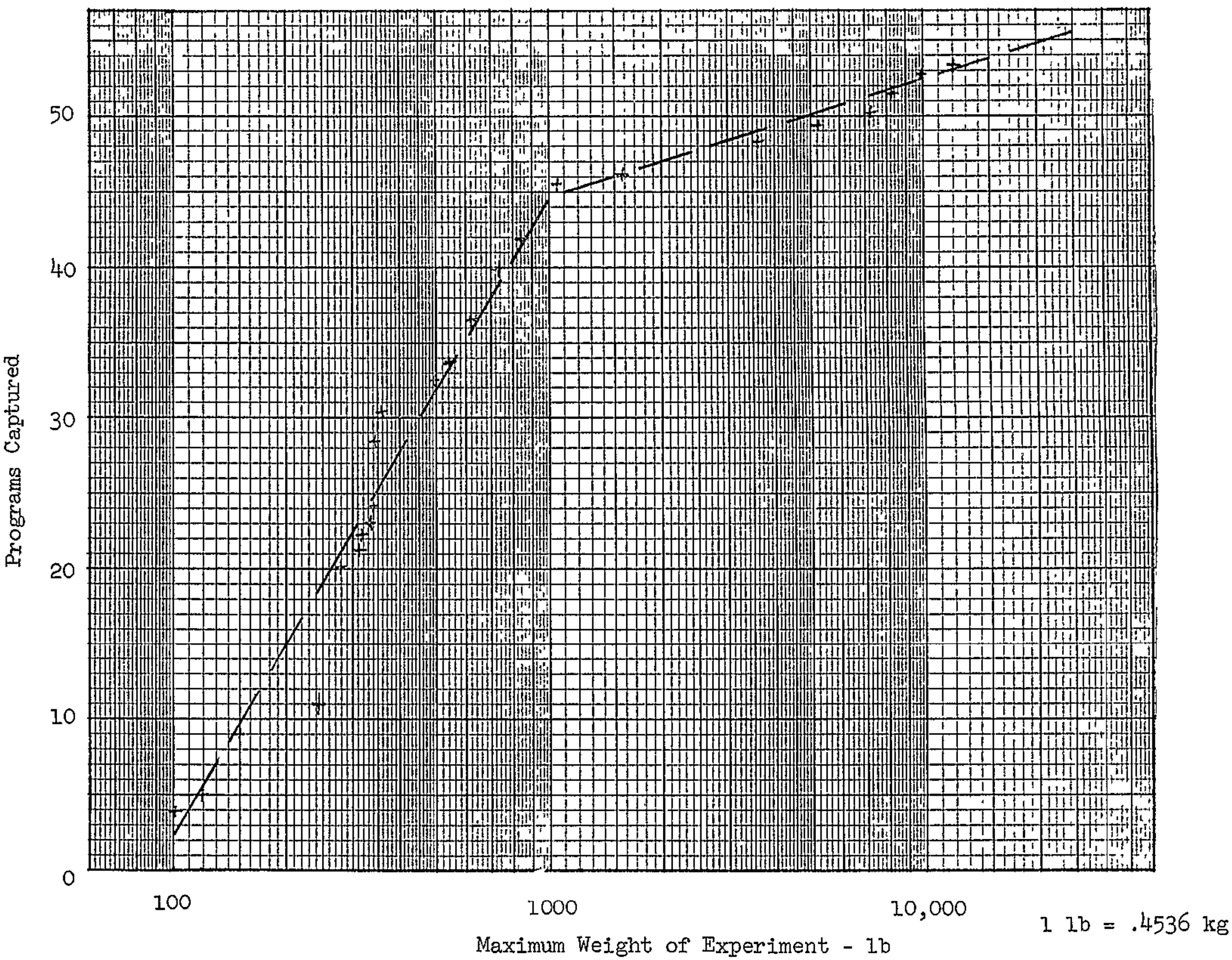


Fig. 7-3 Programs Captured vs Maximum Weight of Experiment

Group	Orbit	Bit Rate	Missions/Flights
Explorers	Low Earth Approx. 20,000 Approx. 1 A.U.	$10^5 - 10^6$ $10^5 - 10^6$ 10^5	2/28 2/22 1/13
Physics	Low Earth Approx. 20,000 Approx. 1 A.U.	$10^4 - 10^5$ $10^4 - 10^5$ $10^4 - 10^5$	1/2 3/6 2/4
Astronomy	Low Earth	$10^5 - 10^6$	4/5
Planetary	"Inter- Planetary"	$10^3 - 10^4$ $> 10^6$	10/18 1/1
Earth Obs.	Low Earth Synchronous	$10^5 - 10^6$ $10^5 - 10^6$	6/62 5/35

Fig. 7-4 Communications Subsystems Requirements

The application of these data to subsystem selection is discussed following.

7.3.3 Subsystems Requirements Selection

The objective of this step was to choose standard requirements options for each subsystem which would approximate the best return on investment in each subsystem sufficiently well to provide a valid demonstration of the economic advantages of developing standard spacecraft and subsystems.

It was assumed that a standard subsystem would be applied to a particular mission only if it would save money to do so. Specifically, it would be applied if the total recurring costs for the mission, using the standard subsystem, would be less than the sum of the non-recurring and the recurring costs for a mission peculiar subsystem. Since the analyses showed that the mission model contained a large number of missions requiring only moderate subsystem performance and relatively few requiring extraordinary performance, care was taken to limit "overkill" and to provide for each subsystem, options which were a reasonably good match to these moderate-performance missions. Analogically, if one had many 10 lb packages to deliver around town, one would buy a panel truck rather than try to use a ten ton diesel truck with lift tail gate.

7.3.3.1 Effects of Spacecraft Weight. While it was not expected that payload weight would be a strong cost driver it was considered desirable to identify the approximate range to be covered before proceeding to select subsystems requirements, as an aid to sizing.

The mission model can be divided into three groups:

(a) Missions calling for small vehicles of the Explorer class, with payloads in the 100-150 lb (45-68 kg) region, which would be fairly simple and could be spin-stabilized

(b) A large group of missions, approximately 35, with payloads in the range 250-1000 lb (113-454 kg) plus one each at 1100 and 1600 lb (500 and 730 kg).

Many of these require 3-axis stabilization and a spacecraft of moderate sophistication

(c) Seven missions requiring payload weight from 3900 lb to 12270 lb (1770 to 5600 kg), some of which are RAM candidate experiments

Group (a) is small but each mission involves a substantial number of flights (13 for each magnetosphere exploration mission, for example). This group would in any case tend to generate its own standard spacecraft. At the other extreme, missions in group (c) might use standard subsystems applicable to group (b), but any standard spacecraft aimed at group (c) would be grossly oversized for most of group (b). The strategy pursued below is, therefore, to concentrate primarily on the high volume of missions in group (b) but to also consider a second spacecraft for group (a).

7.3.3.2 Guidance, Navigation, Stabilization and Control (GNSC). Although the requirements distribution curve for the GNSC subsystem is rather smooth (Fig. 7-1) it appears that the relationship between cost and capability is not, but rather tends to show plateaus associated with the introduction of additional types of components as requirements are made more stringent. At the lowest level it is sufficient to provide a spin-stabilized system with provision for reorientation of the spin axis. The next major step up is to a three-axis stabilized system with a pointing reference accuracy of about 0.2 deg; there appears to be relatively little difference in cost between such a system and one with an accuracy of 2 deg. Such a system has adequate performance for 15 out of 24 missions requiring inertial orientation reference and for 18 out of 26 missions requiring earth-referenced pointing. These groups of missions, which are about 65 percent of the total, contain a substantial number of flights, especially in the earth-oriented group. It is therefore important to provide standard options which are not overpriced in recurring costs for this group.

Capture of the remaining high performance requires the addition of components such as star trackers which are not required at the lower performance levels.

Introduction of these produces a step increase in cost, but does not require a completely new GNSC subsystem lacking commonality with the cruder system. It is therefore considered economically and technologically proper to call for high performance options in both inertially-referenced and earth-referenced applications.

Based on the above reasoning, requirements were selected for five GNSC options, as follows and shown in Fig. 7-5:

a. Spin-stabilized with spin axis reorientation capability

b. 3-axis stabilized, inertially referenced, 0.2 deg pointing reference accuracy

c. 3-axis stabilized, inertially referenced, 15 $\widehat{sec}$ pointing reference accuracy

d. 3-axis stabilized, earth referenced, 0.2 deg pointing reference accuracy

e. 3-axis stabilized, earth referenced, earth pointing reference accuracy, 4 $\widehat{min}$ at 500 nm, 15 $\widehat{sec}$ at synchronous altitude

7.3.3.3 Data Links. Considerations similar to those for the GNSC subsystem apply also to the wide and narrow band data links.

From preliminary discussions with CDPI subsystem specialists, it appeared that a basic unified S-band package, with transmitter/antenna options appropriate to the specific class of application was the most appealing candidate. Using this approach, options were derived appropriate to a spinning satellite and to non-spinning satellites for low-orbit, synchronous orbit, and planetary applications. These options also are summarized in Fig. 7-5.

It may be observed that the main differences between the earth orbital options lies in the selection of an appropriate antenna.

SUBSYSTEM TYPE	ELECTRICAL (BATTERY & SOLAR ARRAY)	GUIDANCE, NAVIGATION, STABILIZATION CONTROL	TELEMETRY (S/C DATA & COMMAND)	COMMUNICATION (EXPERIMENT DATA) (S-BAND*)
A	• 350 W • Sun-Orient. Solar Array	• Stellar/Solar Ref. • 10 Arc Sec • Inertial Platform	• 8-33 BPS • 50 W Xmtr (Ip) • OMNI Antenna	• 10^3-10^4 BPS • 50 W Xmtr (Ip) • 10-30 ft Hi-Gain Tracking Antenna
B	• 700 W • Sun-Or. S/A	• Stellar/Solar Ref. • 15 Arc Min • Inertial Platform	• 10^4 BPS • 2 W Xmtr (SEO) • OMNI Antenna	• 2×10^6 BPS • 2 W Xmtr (SEO) • 3 Ft Fixed Hi-Gain Antenna
C	• 1050 W • Sun-Or. S/A	• Earth Reference • 15 Arc Min • Inertial Platform	• 10^5 BPS • 2 W Xmtr (LEO) • OMNI Antenna	• 10^7 BPS • 2 W Xmtr (SEO) • 6 Ft Hi-Gain Tracking Antenna
D	• 100 W • Body-Mounted Spinning Array	• Spin Stabilized • Axis Orientation Control		• 2×10^7 BPS • 2 W Xmtr (LEO SPIN) • Toroidal Antenna
E		• Earth Reference & Star Tracker • 4 Arc Min • Inertial Platform		• 10^6 BPS • 2 W Xmtr (LEO) • OMNI Antenna
F				

*30 Ft Dia Ground Receiver Antenna, Except for Interplanetary, Which Uses 210 Ft Antenna

1 ft = 0.4536 m

Fig. 7-5 Standard Subsystem Types & Characteristics

It is assumed that a communications satellite will require only a narrow-band data down link.

7.3.3.4 Electrical Power. Selection of standard electrical subsystem options presents a problem different from those of the GNSC and communications subsystems because the cost-versus-capability relationship is, over the range of concern, smoother and without natural plateaus. The greatest volume of missions have requirements in the 100-1000 watt range. Unfortunately, however, the unit cost of a standard 1000 watt system would be too high to be economically acceptable for a mission requiring, for example, 250 watts on each of five flights. It is therefore necessary to select intermediate options below the 1000 watt level.

Considerable exploratory analysis was performed to identify such options which would provide optimum capture of the missions in the model - i.e., the best relationship between investment and saving. There is a small group of missions requiring 100 watts, which will probably have spin-stabilized spacecraft; the bulk of the remaining missions lie in the range from 200 to 1000 watts. This fact, together with some exploratory analyses led to selection of 350 watts as a representative module, leading to options of 350, 700 and 1050 watts which would potentially service 15, 19 and 5 missions, respectively.

7.3.3.5 Environmental Control and Experiments. It is desirable that a standard spacecraft be able to accept any of a wide range of experiments, each of which will have its own, mission-peculiar environmental control requirements. It is therefore proposed that each version of the standard spacecraft provides its own environmental control for everything up to the experiment interface and a basic level of control for the experiment package. Additional requirements would be a mission-peculiar expense.

A standard interface with the experiment should, however, be provided in the areas of attachment pickup points, standardized voltages and connectors, command and data handling interface and similar areas involving the support

of the experiment by the spacecraft. For example, experiment data would be delivered to a standard interface at which it would be sampled, recorded, processed and transmitted to the ground by the spacecraft.

7.3.4 Application of Standard Subsystems and Spacecraft to Mission Model.

The proposed subsystem options were summarized in Fig. 7-5. The applicability of these options to each mission in the mission model is summarized on Figs. 7-6a through 7-6d, together with the quantity of flights in each mission. A summation is presented on Fig. 7-6d of the number of programs and flights to which each spacecraft would be applied: for example, GNSC option "E" applies to 8 missions, involving 75 flights. In this analysis the minimum-performance subsystem which will fulfill mission requirements was chosen. Results were encouraging in that each subsystem operation was found to be applicable to a substantial number of missions.

Integration of various combinations of these subsystems into a standard spacecraft was examined in detail. In summary, it was determined that no single standard spacecraft is effective for a substantial number of missions in the model unless it either:

a. has substantial excess capability for many missions, or

b. provides a number of options for each subsystem.

Either of these approaches will enable substantial mission capture as far as technological capture is concerned. It is, however, believed that (b) is greatly superior to (a) when "economic capture" is considered and will provide a much better return on investment.

These conclusions, together with the first-cut subsystem requirements formed the starting point for the brief subsystem design and costing studies reported in subsection 7.5.

MISSION		ORBIT		NO. OF SPACE-CRAFT	STANDARD SUBSYSTEM TYPE/QTY				
NAME	NO.	ALTITUDE	INCL.		GNSC	COMM	T/M	ELECT	REMARKS
		(N.M.)	PHYSICS AND ASTRONOMY						
Astronomy Explorers - A	1	270/270	28.5°	15	D 15	D 15	C 15	D 15	
- B	2	Sync	0°	9	D 9	B 9	B 9	D 9	
Magnetosph. Exp. (Low)	3	2000/100	28.5° & 90°	13	D 13	D 13	C 13	D 13	
(Medium)	4	60,000/100	28.5° & 90°	13	D 13	D 13	B 13	D 13	
(High)	5	1 A.U. Helio	Ecliptic	13	D 13	--	--	A 13	
OSO	6	350/350	Any	1	A 1	E 1	C 1	A 1	
Gravity/Relativity A, C, E	7	300/300	85° - 95°	2	B 2	E 2	C 2	B 2	
B, D	8	1 A.U. Helio		2	B 2	A 2	A 2	A 2	
Radio Interferometer	9	40,000/40,000	28.5°	2	A 2	C 2	B 2	A 2	
Solar Orbit Pair Sync	10	Sync	30°	2	B 2	B 2	B 2	A 2	5 Yr Life
1 A.U.	11	1 A.U. Helio	Ecliptic	2	B 2	A 2	A 2	--	5 Yr Life 1.5 kw
Optic. Interf. Pair	12	Sync	30°	2	A 2	B 2	B 2	A 2	
High Energy Astro Obs.	13	230/230	30°	2	A 2	E 2	C 2	A 2	12,270 lb Expt.
Large Stellar Tel.	15	230/230	28.5°	1	A 1	E 1	C 1	--	1.5 kw 8250 lb Expt.
Large Solar Obs.	17	350/350	30°	1	A 1	E 1	C 1	C 1	7520 lb Expt.
Large Radio Obs.	19	350/350	30°	1	A 1	E 1	C 1	--	2 kw 10,000 lb Expt.

Fig. 7-6a Applicability of Subsystems Options to Missions

1 lb = .4536 kg
1 nm = 1.852 km

MISSION		ORBIT		NO. OF SPACE-CRAFT	STANDARD SUBSYSTEM TYPE/QTY				
NAME	NO.	ALTITUDE (N.M.)	INCL.		GNSC	COMM	T/M	ELECT	REMARKS
			NASA EARTH OBSERVATION						
Polar EOS	21	500/500	SS	12	E 12	E 12	C 12	B 12	
Sync EOS	22	Sync	0°	6	E 6	C 6	B 6	B 6	
Earth Physics	23	400/400	90°	7	E 7	C 7	C 7	A 7	
Sync Met	24	Sync	0°	2	C 2	C 2	B 2	A 2	
Tiros	25	700/700	SS	3	C 3	B 3	C 3	A 3	
Polar ERS	26	500/500	SS	6	E 6	E 6	C 6	B 6	
Sync ERS	27	Sync	0°	7	E 7	C 7	B 7	B 7	
			NASA COMMUNICATION/NAVIGATION						
ATS	28	Sync	0°	7	E 7	C 7	B 7	--	8 kw
Small ATS Sync	29	Sync	0°	12	C 12	C 12	B 12	B 12	
Low	30	3000/300	90°	12	C 12	E 12	C 12	B 12	
Co-Op ATS Sync	31	Sync	0°	2	C 2	C 2	B 2	B 2	
Low	32	3000/300	90°	2	C 2	E 2	C 2	B 2	
Medical Net	33	Sync	0°	2	C 2	Comm. part of P/L	B 2	C 2	
Educ. Broadcast	34	Sync	0°	2	C 2		B 2	--	2 kw
Follow-on Sys.Demo.	35	Sync	0°	20	C 20		B 20	C 20	
Tracking/Data Relay	36	Sync	0°	10	C 10	C 10	B 10	B 10	
Planetary Relay	37	Sync	0°	9	C 9	C 9	B 9	A 9	

Fig. 7-6b Applicability of Subsystem Options to Missions

1 nm = 1.852 km

MISSION		ORBIT		NO. OF SPACE-CRAFT	STANDARD SUBSYSTEM TYPE/QTY				
NAME	NO.	ALTITUDE	INCL.		GNSC	COMM	T/M	ELECT	REMARKS
			PLANETARY						
Mars Viking	50			2	B 2	A 2	A 2	C 2	
Mars Sample Ret'n	51			2	B 2	A 2	A 2	--	15 kw
Venus Explorer	52			1	B 1	A 1	A 1	A 1	
Venus Radar Map	53			1	A 1	A 1	A 1	C 1	
Venus Expl./Lndr.	54			2	B 2	A 2	A 2	B 2	
Jup. Pioneer/ Orbiter	55			2	B 2	A 2	A 2	--	2 Yr
J.U.N. Grand Tour	56			2	A 2	A 2	A 2	--	9 Yr long distance from sun
Jup. Tops Orb/ Probe	57			2	B 2	A 2	A 2	--	3 Yr
Uran. Tops Orb/ Probe	58			2	B 2	A 2	A 2	--	7 Yr
Asteroid Survey	59	3 A.U.	Helio	1	B 1	A 1	A 1	--	4 Yr 6 kw elec.prop.
Comet (Halley) Rend'v's	60			2	B 2	A 2	A 2	--	3 Yr long dist. from sun

Fig. 7-6c Applicability of Subsystem Options to Missions

MISSION		ORBIT		NO. OF SPACE-CRAFT	STANDARD SUBSYSTEM TYPE/QTY				
NAME	NO.	ALTITUDE (N.M.)	INCL.		GNSC	COMM	T/M	ELECT	REMARKS
			NON-NASA						
COMSATS	70	Sync	0^{o}	11	C 11	Communications Part of Payload	B 11	C 11	
U.S. Dom. COMSATS	71	Sync	0^{o}	22	C 22		B 22	--	2 kw
Foreign Dom. COM-SATS	72	Sync	0^{o} - 28^{o}	26	C 26		B 26	B 26	
Nav/Traffic Cont.	73	30,000/16,000	29^{o}	8	C 8		B 8	A 8	
Nav/Traffic Cont.	74	Sync	5^{o}	8	C 8		B 8	A 8	
TOS MET	75	700/700	SS	12	C 12	E 12	C 12	A 12	
Sync MET	76	Sync	0^{o}	12	C 12	C 12	B 12	A 12	
Polar ER	77	500/500	SS	22	E 22	E 22	C 22	B 22	
Sync ER	78	Sync	0^{o}	8	E 8	C 8	B 8	B 8	

SUMMATION: SUBSYSTEMS TOTALS (MISSIONS/SPACECRAFT)

GNSC	COMM	T/M	ELECT
A(9/13)	A(13/23)	A(13/23)	A(16/86)
B(13/24)	B(4/16)	B(23/202)	B(14/129)
C(18/175)	C(12/84)	C(16/112)	C(6/37)
D(5/63)	D(3/41)		D(4/50)
E(8/75)	E(12/74)		

1 nm = 1.852 km

Fig. 7-6d Applicability of Subsystem Options to Missions

7.4 DESIGN APPROACHES FOR STANDARDIZED SPACECRAFT AND SUBSYSTEMS

This subsection describes the individual approaches to subsystem standardization. In each case the problem of how to satisfy the various mission requirements is analyzed and design options are detailed. Limitations of the selected approaches are discussed, as are suggested alternatives or areas for future study. The impact on spacecraft development, manufacturing, testing, and operations is indicated.

The standard subsystem designs are characterized by a small number of subsystem variants or options to satisfy a wide spectrum of mission requirements. Although these standard options will, in many cases, have capabilities in excess of mission requirements, a net cost saving will result by eliminating the expense of repetitive developments of optimized, program-peculiar equipment.

7.4.1 Summary of Performance Options

- Electrical Power (3 options)

  350 watts max.
  700 watts
  1050 watts

- Communications and Data Processing (10 options)

$$\left.\begin{array}{l} 3 \times 10^5 \text{ bps} \\ 4 \times 10^6 \text{ "} \\ 1.6 \times 10^7 \text{ "} \end{array}\right\} \text{Low Earth Orbit}$$

$$\left.\begin{array}{l} 3 \times 10^3 \text{ bps} \\ 3 \times 10^4 \text{ "} \\ 10^5 \text{ "} \\ 10^6 \text{ "} \\ 10^7 \text{ "} \end{array}\right\} \text{High Earth Orbit}$$

10^5 bps   1 AU Interplanetary

10^5 bps   3 AU Interplanetary

- Attitude Control/Stationkeeping Propulsion (2 options)

  8,000 lb-sec and 0.5 lbf (35,600 Newton-sec and 2.22 Newton)

  30,000 lb-sec and 5.0 lbf (133,400 Newton-sec and 22.2 Newton)

- Stabilization and Control (5 options)

  $\left.\begin{array}{l} 20\ \widehat{\text{sec}} \\ 15\ \widehat{\text{min}} \end{array}\right\}$ Stellar Orientation

  $\left.\begin{array}{l} 25\ \widehat{\text{sec}} \\ 4\ \widehat{\text{min}} \\ 15\ \widehat{\text{min}} \end{array}\right\}$ Earth Orientation

These options represent the lowest number of divisions of the requirements which are believed desirable for a 3-axis stabilized standard spacecraft. However, other optional capabilities lying between any two listed can be realized by adding or deleting a module or components within a standard subsystem module.

7.4.2 Approaches and Groundrules for Standard Subsystem Design

7.4.2.1 Choice of the Level of Standardization. When spacecraft standardization is pursued with the objective of economic savings, the first question is at which systems level to implement standardization to effect the greatest dollar benefit. The level must be low enough so that as many as possible of normally repetitive hardware development/test efforts can be replaced by a fewer number. On the other hand, the level must be high enough so that as much as possible of repetitive elements of systems integration and systems test can be eliminated.

For the purpose of the mission model capture analysis it was assumed that standardization would occur at the subsystems level. Concerning a choice

of the proper level, it was found that the spectrum could actually be extended in both directions without contradiction:

- in the lower direction by utilizing a modular build-up of subsystems capability, and
- in the upward direction by the definition of standard interfaces.

7.4.2.2 Standard Spacecraft/Subsystem Approach. The Standard Spacecraft concept exploits transportation economics of the Space Shuttle to launch payloads which are relatively free of size and weight constraints. With these constraints removed, it is feasible to use a small number of "standardized" subsystem designs to satisfy a majority of the future missions and thereby provide considerable cost savings through elimination of the normal specialized-development program which optimize equipment design for each mission's spacecraft.

The Standard Spacecraft approach will provide subsystem designs which in many cases will exceed the mission requirement but will be less costly than a specially-tailored design. The actual cost savings will depend upon the number of flights involved for the particular mission.

7.4.2.3 Groundrules for Standard Subsystems. The groundrules for Standard Spacecraft Subsystems were established:

- Space Shuttle/Space Tug-launched
- Mission model requirements for programs in the 1978-1990 time period
- 1970-1975 technology (concepts reduced to practice)
- Two-year life without refurbishment; up to three reuses
- Subsystem reliability goal 0.80 to 0.85
- Low-cost: Trade weight increase for cost reduction
- Minimize number of variants: Favor over-design over a multiplicity of mission-peculiar parts

- Use modular concepts for flexibility in system implementation and for on-orbit repair or replacement

7.4.3 Standard Spacecraft Functional Interfaces

An intrinsic part of the standard subsystem designs is the grouping of equipment into modules. This characteristic not only facilitates repair and refurbishment, it also permits build-up to desired system capability levels. The modules would be designed to be interchangeable (with an identical module) at the interface connections without need for module or spacecraft re-calibration. The following paragraphs lists all significant inter-subsystem interfaces; these interfaces constrain both the spacecraft design and the module groupings.

7.4.3.1 Active Interfaces (Power and Signal).

a. Electrical: (EPS)

(1) Power Distribution TO all Subsystems:

- 28 VDC unreg. bus
- Optional 28 VDC reg. bus
- Optional 115 VAC/400HZ Bus

b. Stabilization & Control: (S&C)

(1) Actuation Signals TO attitude control propulsion system via digital drive electronics

(2) Attitude error signals FROM payload experiment sensors (for very high precision pointing only, that is less than 20 arc sec)

c. Telemetry, Instrumentation:

(1) Status signals FROM all subsystems to data distribution

(2) Interface unit for transmission or storage

d. Test:

(1) Status signals FROM all subsystems to space shuttle/tug for readiness checks (same as "Telemetry")

(2) Activation/stimulus signals TO all subsystems from shuttle/tug for readiness checks and activation

e. Commands: (1) Real time and stored actuation commands TO all subsystems via DDIU (same as "Test")

f. Communications: (1) Ground - space and space - ground links for status telemetry tracking aids, commands and experiment data transmission

7.4.3.2 Passive Interfaces (Geometric, Kinematic)

a. Structural/Thermal: (1) Establish/maintain S&C inter-sensor alignment; communications antennas - S&C alignment; and experiment sensors - S&C alignment

b. Structural Arrangement: (1) Solar arrays/antennas define solar and gravity gradient disturbance torque environment for S&C/attitude control sizing

(2) Solar arrays/antennas placement compatible with S&C sensor FOV demands.

(3) Modules/module installation for astronaut and/or telefactor handling

7.4.3.3 Integration Interfaces

a. On-Board Computer Software

b. Digital Data Bus/Localized Operations

c. On-board Fault Isolation

7.4.3.4 Spacecraft/Experiment Interfaces. The mission spectrum can be categorized by five payload types:

(1) Astronomical Experiment

(2) Space Physics Experiment

(3) Communications and Applied Technology

(4) Planetary Exploration

(5) Observatory

The viability of the standard spacecraft concept will, to a large degree, depend upon the ease with which the diverse experiments can be mated to the spacecraft proper. The key to this integration lies in successfully defining standardized interfaces which still permit both the experimenter and the payload designer the maximum latitude. The experiment requirements affecting the interface for which specifications must be drawn can be classified as follows:

(1) Loads and Dynamics

(2) Electric Power

(3) Stabilization and Control

(4) Commands In

(5) Data Out

(6) Thermal

Also to be considered are unwanted interactions which must be constrained. Principal items in this area are:

(1) Electromagnetic Interference

(2) Fields-of-View including Spurious Reflection

(3) Outgassing and Thruster Exhaust Products

(4) Elastic Motions

7.4.4 Standard Spacecraft Design and Integration

7.4.4.1 General Configuration. The Standard Spacecraft has been concepted as a compartmentized ("eggcrate" type) structure sized to accept the various

modules of the several subsystems. This structure also includes provisions for attaching an experiment package as may be required to accomplish the various missions appropriate to a Standard Spacecraft. Figure 7-7 is a general arrangement drawing of the Standard Spacecraft.

Four (4) lug type fittings are provided as a part of the spacecraft upper bulkhead, positioned as required for attachment of the experiment packages. The upper bulkhead will contain electrical connectors required for electrical interconnect between the experiment packages and the spacecraft subsystems.

7.4.4.2 Integration of Standard Spacecraft with Transportation Systems. The Standard Spacecraft will be supported in the Space Shuttle cargo bay by means of hold-down and deployment gear designed to secure the payload in the cargo compartment. Four fixed probe devices shown in Figure 7-7 are to be used in cases where spacecraft deployment and retrieval is to be accomplished by means of four (4) Storable Tubular Extendable Members (STEM) and an appropriate cradle. This deployment and retrieval configuration is depicted in Figure 7-8. In the event the Standard Spacecraft is to be used in conjunction with a Space Tug or other boosters, the optional docking ring can be installed in place of the four fixed probes. The docking ring is equipped with appropriate probe and drogue devices. The ring is attached to the spacecraft lower bulkhead by means of four double acting shock mitigation devices.

7.4.4.3 Modular Equipment Installation. Arrangement of subsystem equipment modules is shown in Figure 7-9. Module location is limited by requirements for appropriate view angles, vehicle center of gravity location, alignment accuracy and other similarly important reasons.

7.4.4.4 Structures and Mechanisms. The Standard Spacecraft Structure is a compartmentized aluminum sheet metal structure reinforced or stiffened by extruded aluminum angle and "Tee" structural shapes. Each compartment is provided with guide rails to aid in the rapid installation and removal of the several modules. At the back of each compartment is an electrical connector providing the appropriate module/spacecraft electrical interconnect.

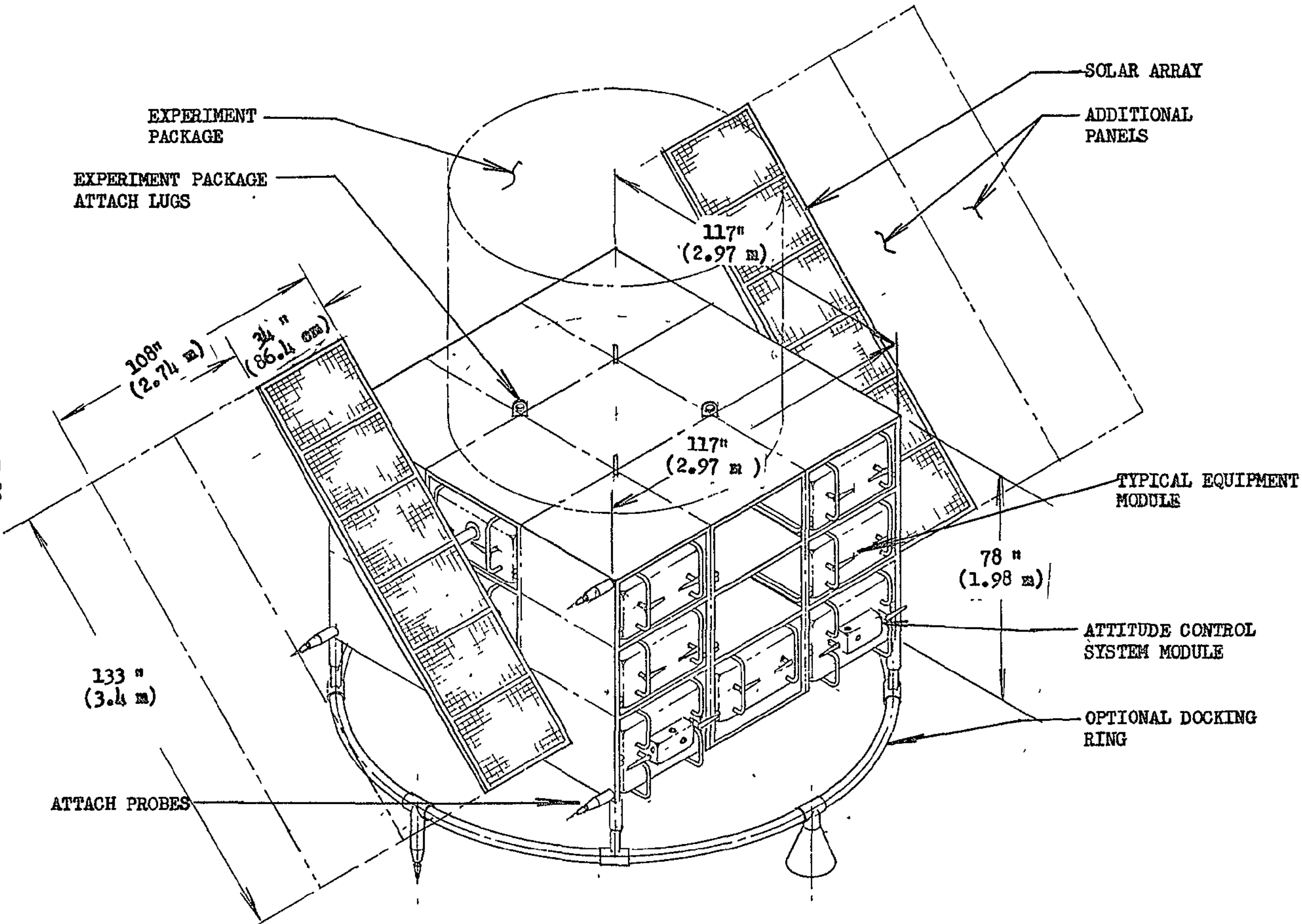


Fig. 7-7 Standard Spacecraft General Arrangement

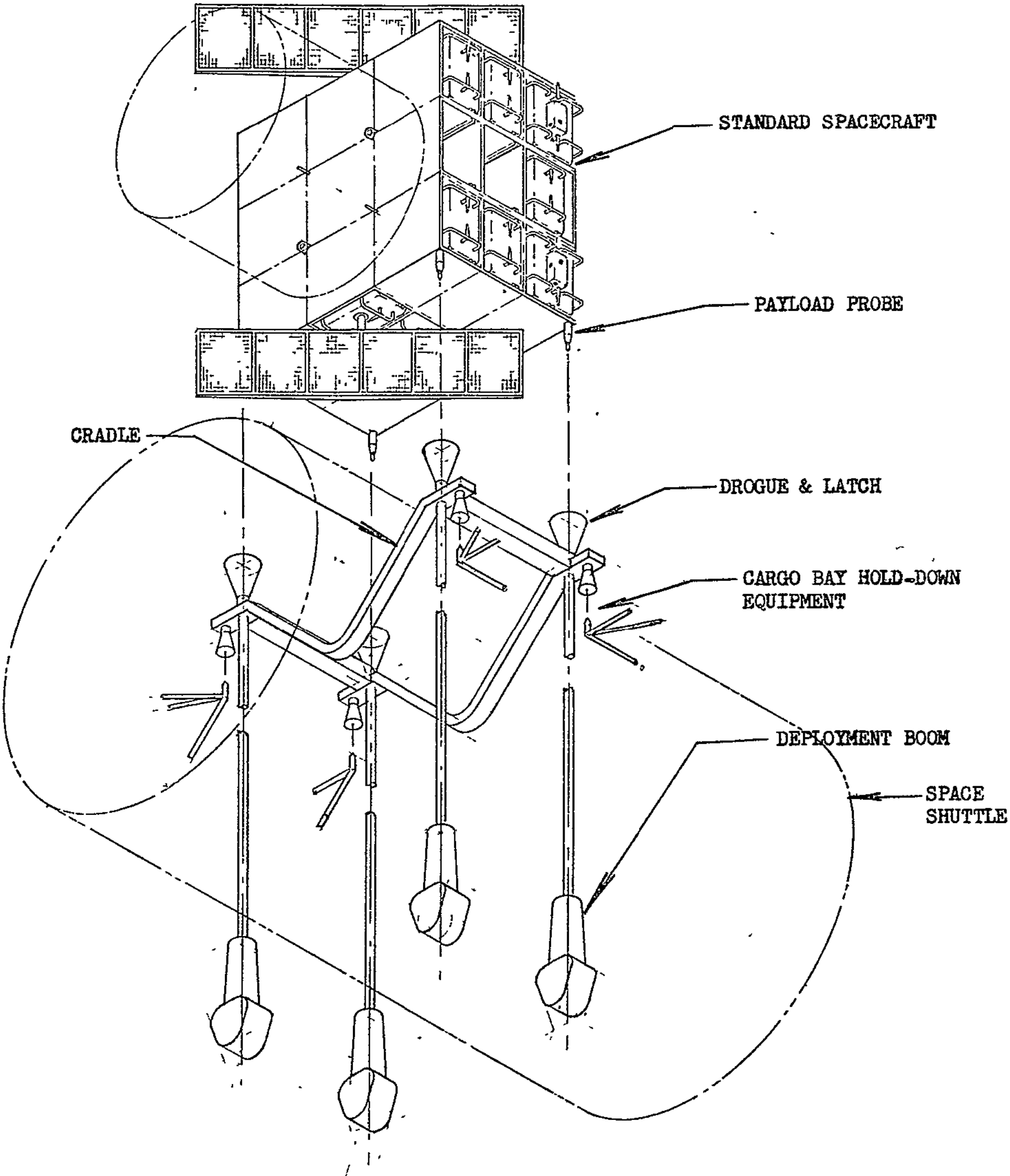
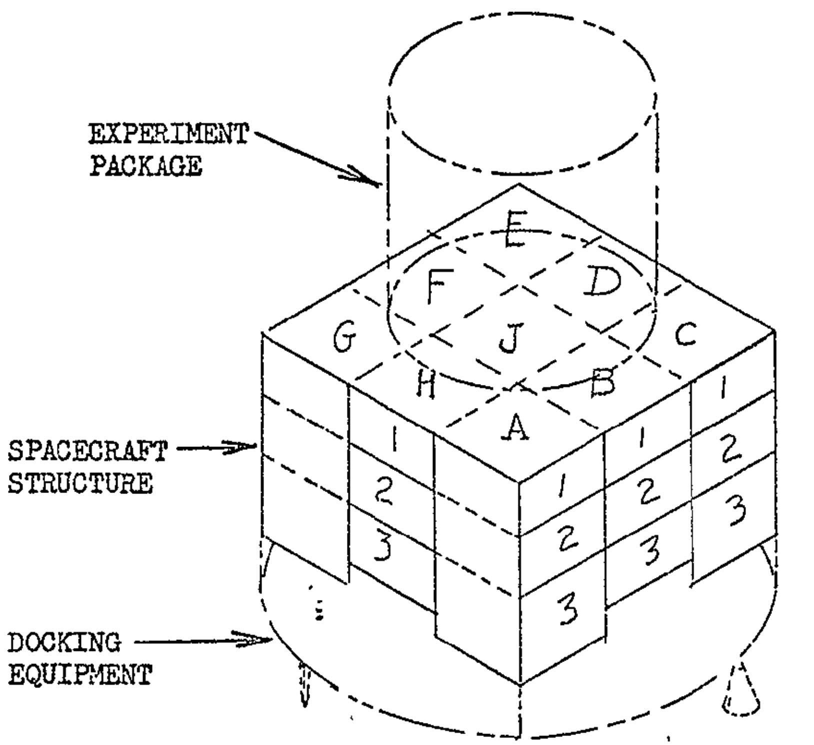


Fig. 7-8 Standard Spacecraft/Shuttle Hold-Down & Deployment Provisions



MODULE COMPARTMENT	
A-1	ATTITUDE SENSING MODULE NO.1
A-2	SPARE
A-3	ATTITUDE CONTROL SYSTEM MODULE NO.1
B-1	COMMAND,DATA PROCESSING & INSTRUMENTATION MODULE NO.1
B-2	ATTITUDE COMPUTATION MODULE
B-3	RESERVED - FOR SOLAR ARRAY TRACKING AND POWER TRANSFER MODULE
C-1	ATTITUDE SENSING MODULE NO.2
C-2	SPARE
C-3	ATTITUDE CONTROL SYSTEM MODULE NO.2
D-1	SOLAR ARRAY TRACKING AND POWER TRANSFER MODULE
D-2	SPARE
D-3	BATTERY MODULE NO. 1
E-1	ATTITUDE SENSING MODULE NO.3
E-2	SPARE
E-3	ATTITUDE CONTROL SYSTEM MODULE NO.3
F-1	COMMAND, DATA PROCESSING & INSTRUMENTATION MODULE NO.2
F-2	REACTION WHEEL MODULE
F-3	RESERVED-FOR SOLAR ARRAY TRACKING AND POWER TRANSFER MODULE
G-1	ATTITUDE SENSING MODULE NO.4
G-2	SPARE
G-3	ATTITUDE CONTROL SYSTEM MODULE NO. 4
J-1	EMPTY - EXCEPT FOR ELECTRICAL INTERCONNECT CABLES
J-2	" " " " " "
J-3	RESERVED - FOR ATTITUDE SENSING MODULE NO. 5

Fig. 7-9 Standard Spacecraft Module Location Diagram

Two fittings are also included in the compartment aft wall, which engage the double acting cams at the back of each equipment module providing force required to connect and disconnect the connector as well as to lock the module in the desired position required for operating purposes. Figure 7-10 depicts the typical equipment module compartment. In some instances module guide rails will require special location dictated by module requirements such as symmetry about the spacecraft centerline.

Standard spacecraft subsystem equipment is housed in specially designed modules that may be easily installed or removed from the spacecraft structure by astronauts. The modules are designed to protect subsystem equipment during ground handling as well as during operational use. Insulation, paint patterns, heaters or other devices for thermal environment control will be included as a part of the module.

Except for the attitude control and solar array drive modules all other standard spacecraft modules are of a standard size and design. Figure 7-11 is a view of the typical module. The module structure is sheet metal reinforced as required to provide strength and rigidity to withstand loads and stresses resulting from module handling and spacecraft operations.

The module is provided with hand rails that will facilitate module handling by the shuttle crew. The base has two "module support rails" which engage the "module guide rails" that are attached to the spacecraft structure.

Rotation of the handles on the face of the model actuate double acting cam devices located on the back face of the module (see Figure 7-12). Rotation of these cam devices, acting in combination with mating fittings attached to the spacecraft structure, provide force required to connect or disconnect the back mounted electrical connector as well as forcing the module positioning lugs into proper engagement with the module guide rails.

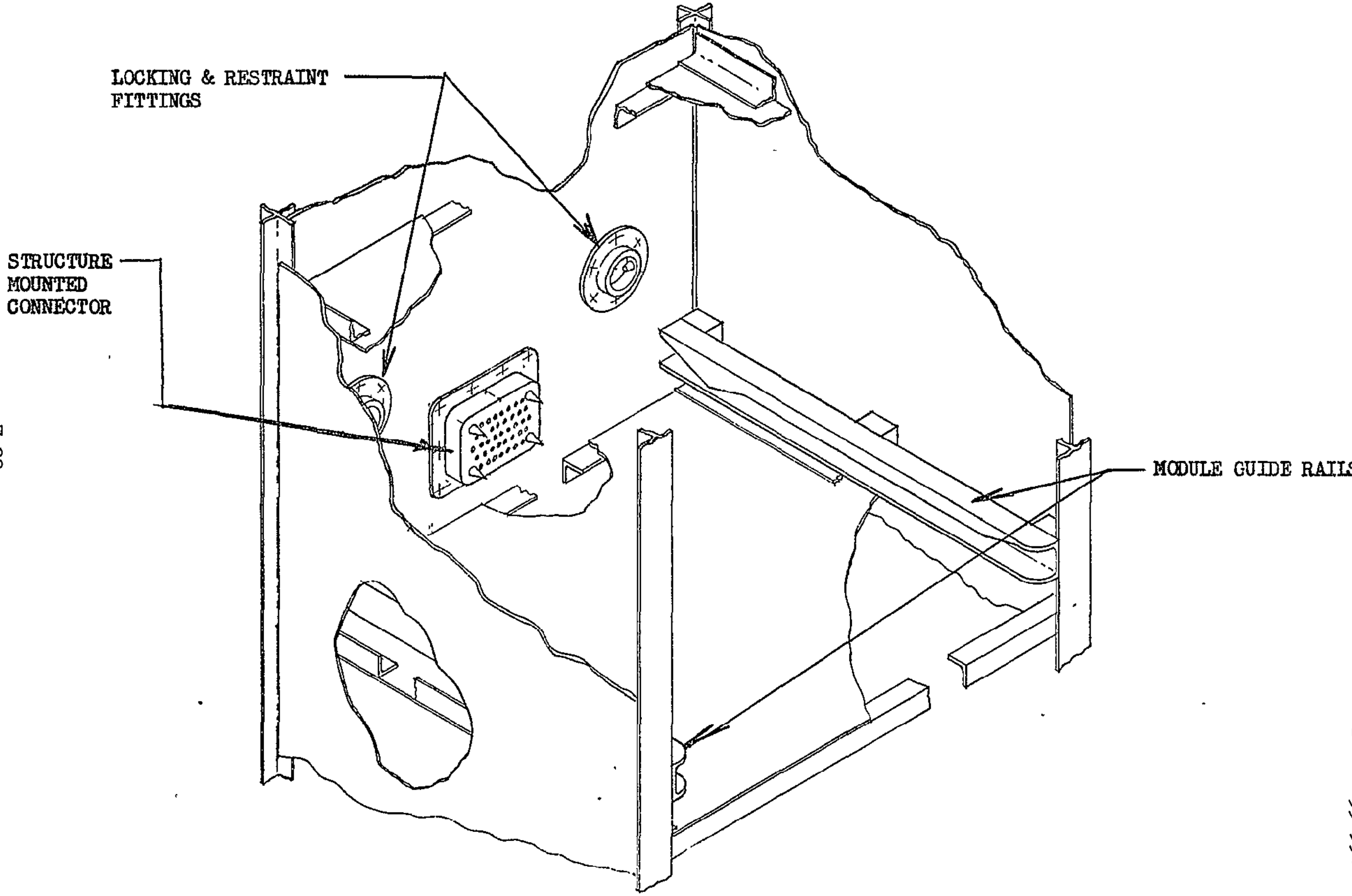


Fig. 7-10 Standard Spacecraft Equipment Module Compartment

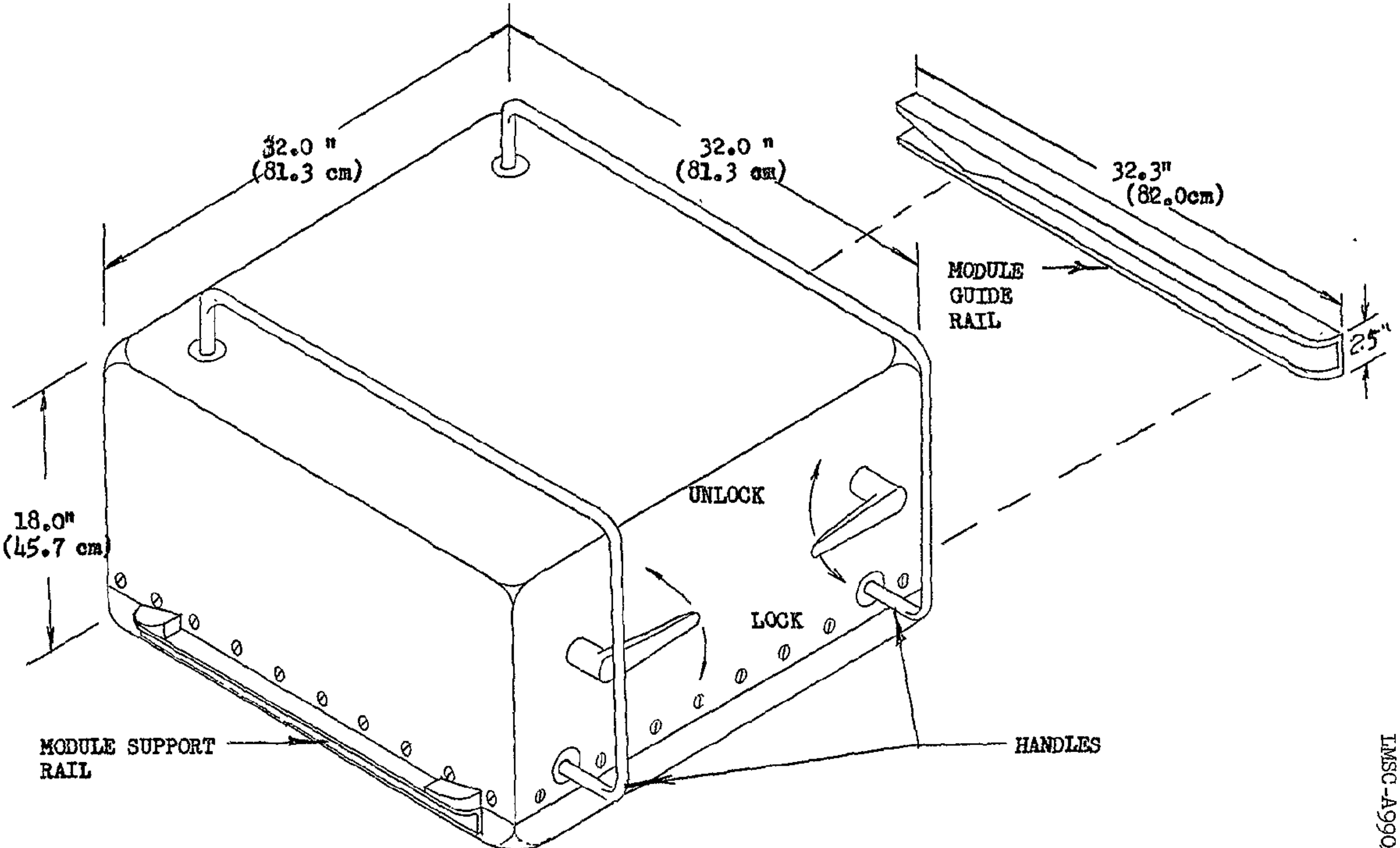
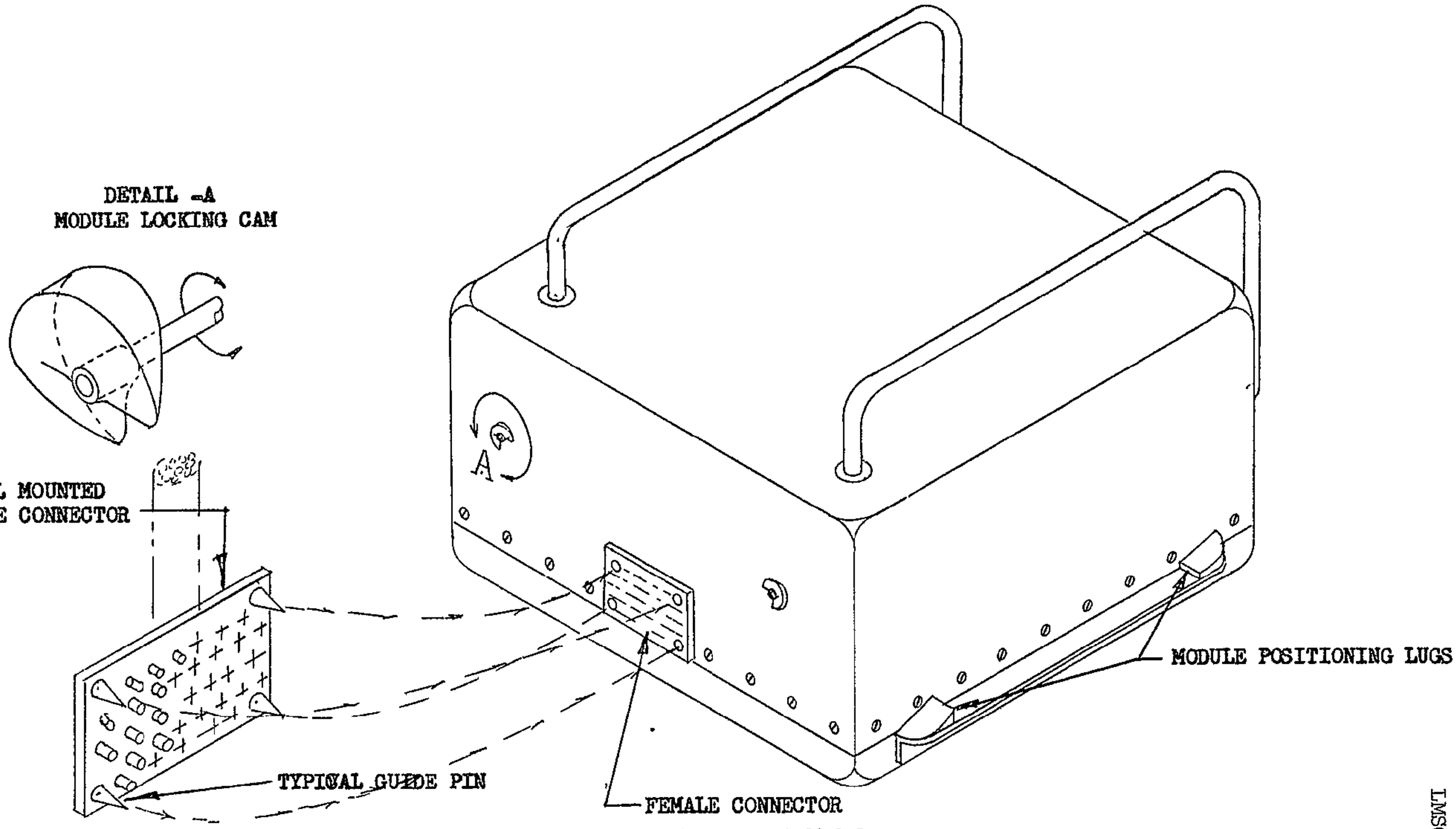


Fig. 7-11 Typical Standard Spacecraft Module



Standard Spacecraft Module

Fig. 7-12 Positioning and Locking Provisions

The Attitude Control System Module, Figure 7-13, is an example of a special module, whose operating function requires particular design features. This module is designed to be externally symmetrical about the center plane that includes the module support rail. This permits use of this module in any of the four corners of the spacecraft by merely inverting the module.

7.4.5 Standard Electrical Power Subsystem (EPS)

7.4.5.1 Electrical Power Requirements. The NASA mission model contains both actuator-stabilized and spin-stabilized spacecraft in sufficient numbers to warrant standardization of their electrical power subsystems. Based on the incidence of requirements (see subsection 7.3), the following parameter variants were selected:

Option	Power (watts)	
A	100 to 350	Actively-stabilized spacecraft
B	350 to 700	
C	700 to 1050	
D	20 to 100	Spin-stabilized spacecraft

7.4.5.2 EPS Design Approaches - Actuator-Stabilized Spacecraft. The earth orbit solar incidence β-angles and inclination angles are essentially unlimited and all values are assumed to be required. The solar array areas associated with these power levels are shown on Fig. 7-14.

The increments of array wing sizes are based on array area rather than spacecraft average load since the β-angle in the low to medium earth orbit can cause a variation by a factor of up to two in array size required for a given average power requirement. The upper ends of the power ranges correspond to little or no shadow periods during operation. The standard EPS options are configured in a modular fashion as can be seen on Fig. 7-15.

a. Solar Arrays. All three array sizes employ two wings on individual shafts

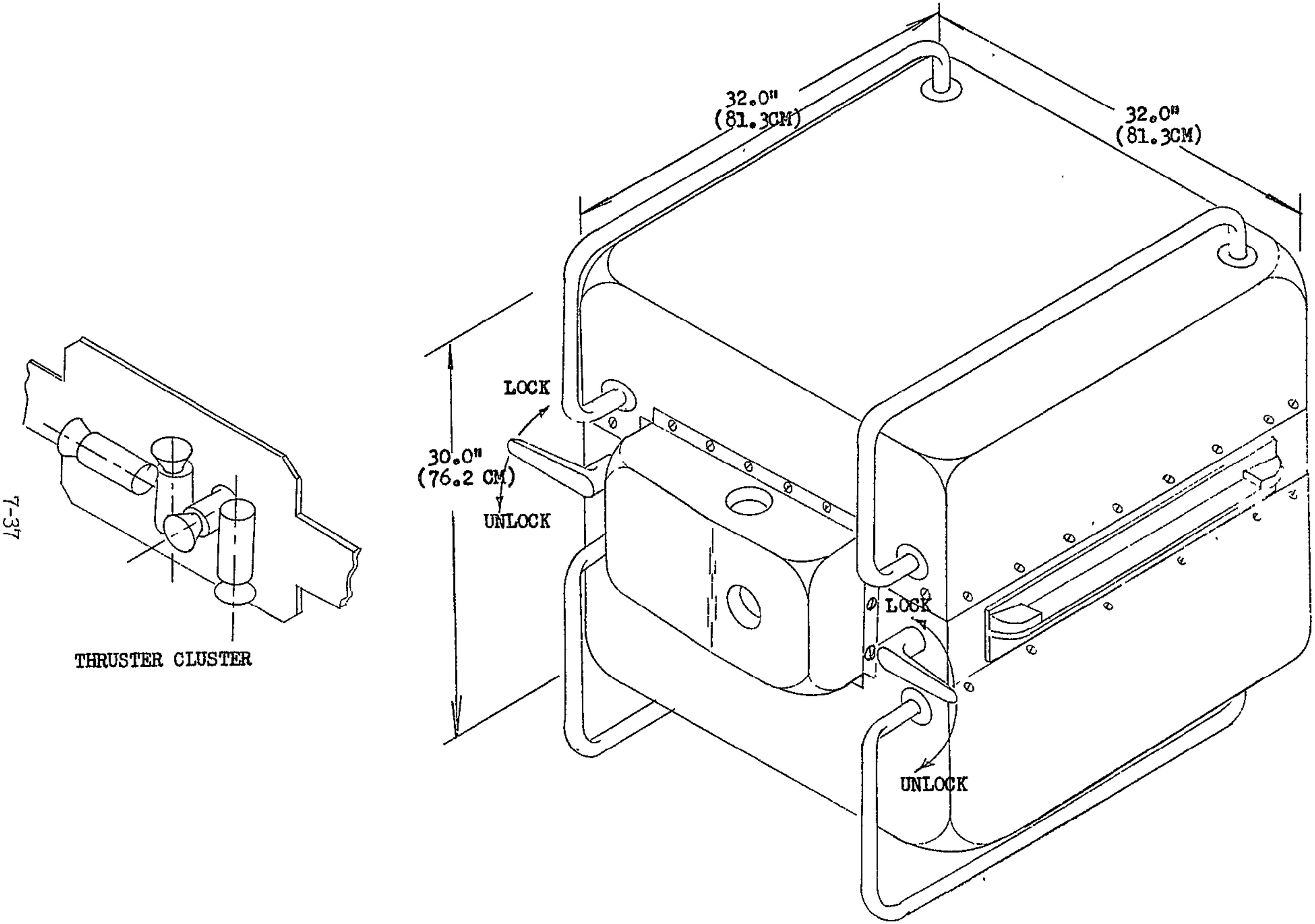


Fig. 7-13 Standard Spacecraft - Attitude Control System Module

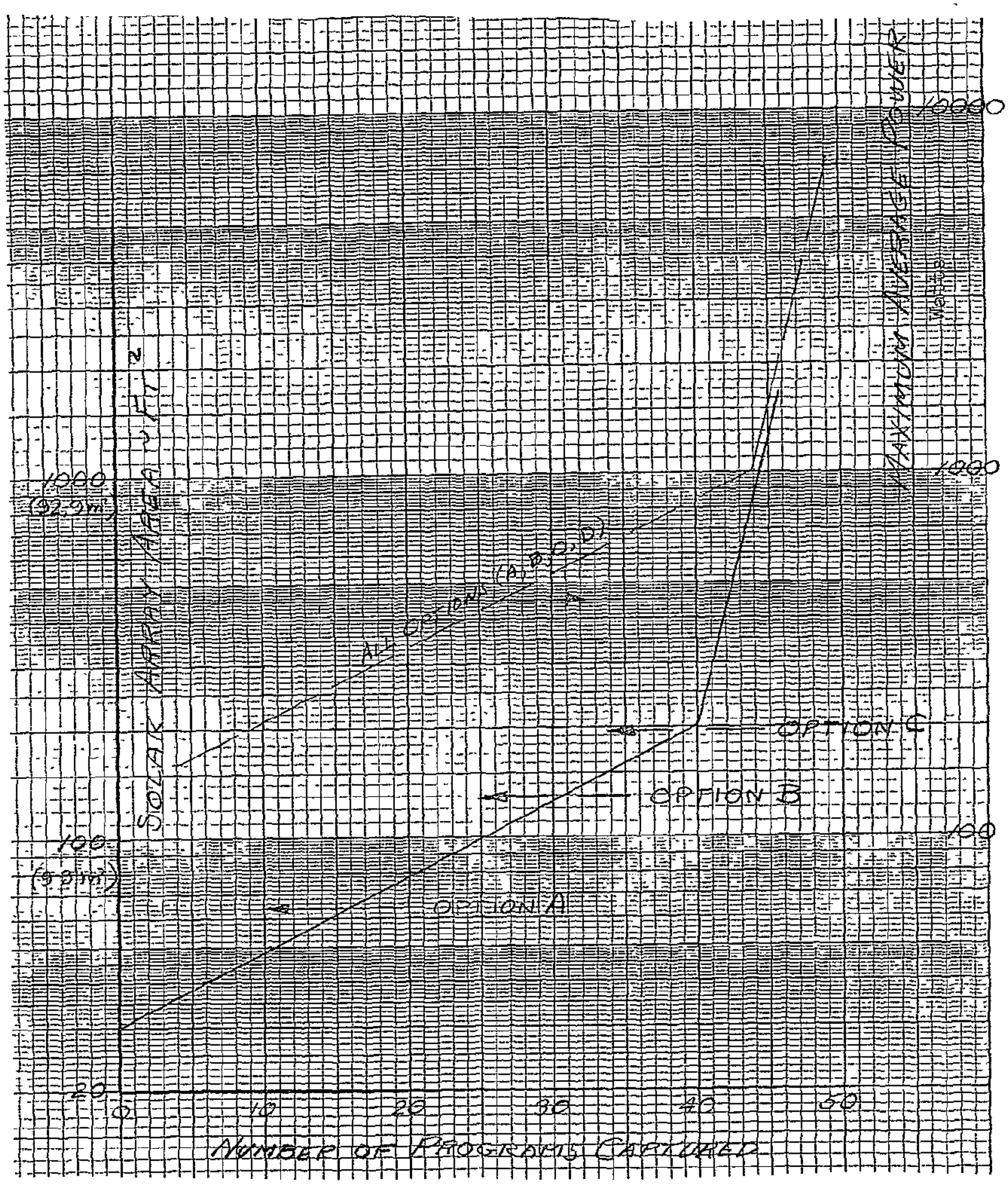


Fig. 7-14 Power Requirements

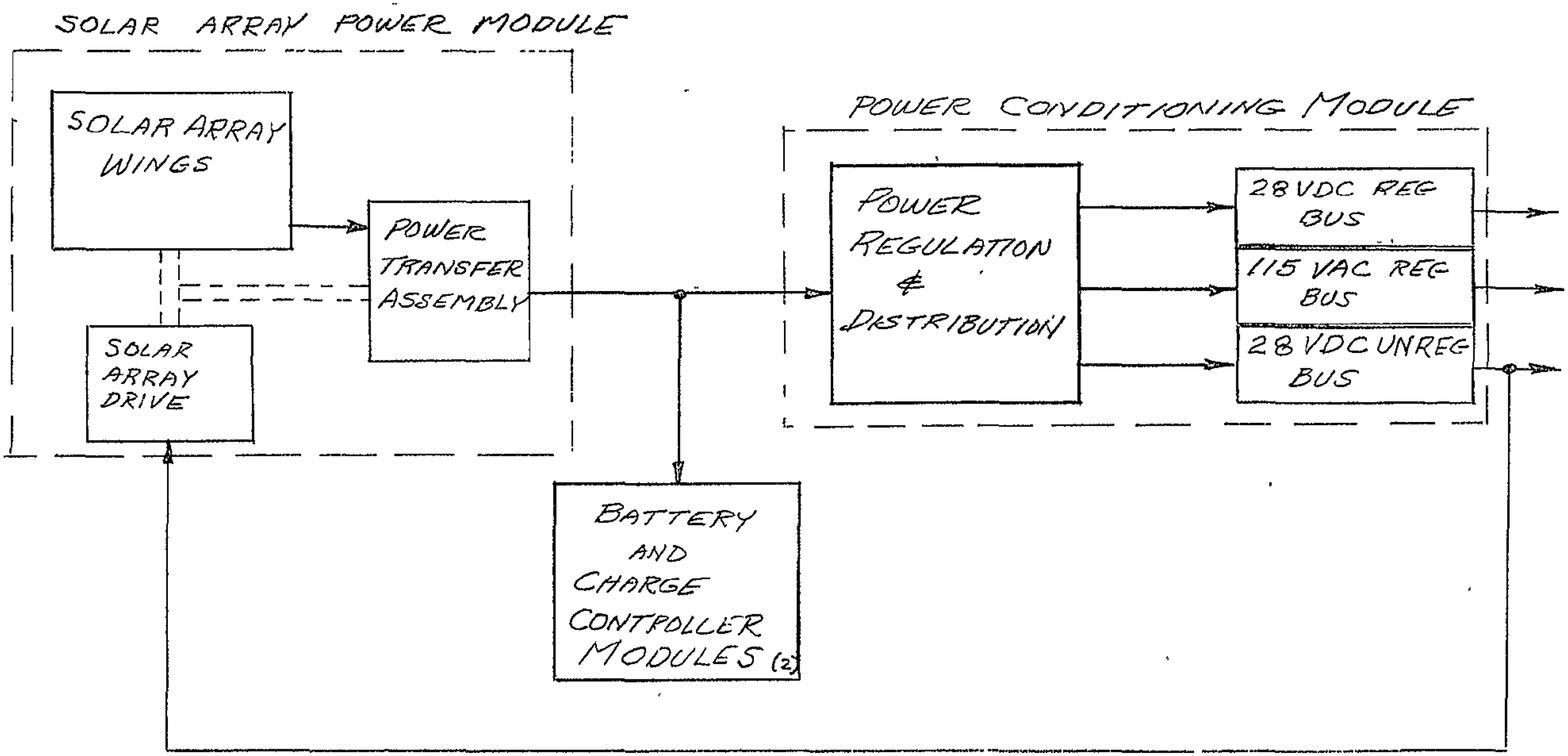


Fig. 7-15 Standard Electrical Power Subsystem Modules and Interconnections

with single axis tracking and slip-ring power transfer provided on each shaft. Investigation of "all-β" orbits indicates that the use of a single axis of the tracking (continuous drive) solar array would require a solar array no more than 20 percent larger in comparison to the size for two-axis orientation.

The tracking units and power transfer assembly are sized for the largest subsystem and are used for all array sizes.

The two array wings are composed of modular sections up to 3 in number as shown on Fig. 7-16. The high length-to-width shape of the sections is selected to allow the first inboard section to be in the on-orbit position in the Shuttle P.L bay when the spacecraft width allows. If additional sections are flown they are folded onto the inboard sections for launch.

The two wings are covered on one side by 12-mil (0.3mm) 2 x 4 cm wraparound contact silicon solar cells with 20-mil (0.51mm) cover-glass pre-attached to reduce the number of cell handling and cleanup operations.

Material costs and manufacturing costs are minimized at the expense of weight in the selected solar cell and coverglass design thicknesses. Smaller thicknesses results in lower yield due to handling losses and larger thicknesses result in higher material costs.

The cost per watt for solar cells is further reduced at the expense of weight increase by lowering the minimum allowable cell power requirements in accepting cells from production run.

The electrical and mechanical configuration of the cells will be selected to provide one panel design which is then repeated to generate the total array. power requirement. The use of a standard panel design for Subsystem variants A, B, and C, using a standard number of solar cells in series will result in the same panel voltage when the cells operate at the same equilibrium temper-

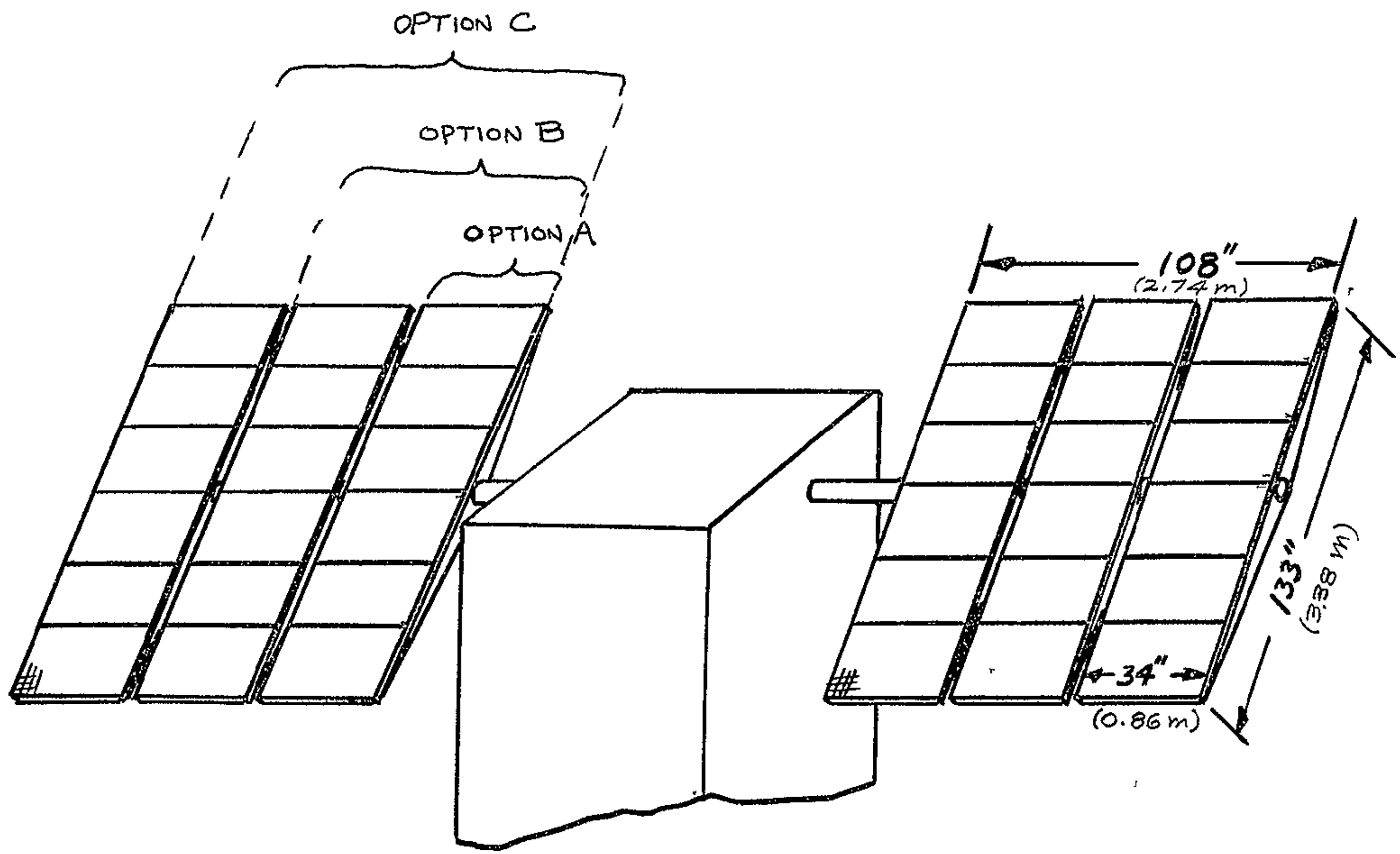


Fig. 7-16 Solar Array Configurations

atures. This temperature condition will not be obtained precisely in all missions with various orbit parameters. The voltage variations, however, should be within acceptable limits for the power conditioning equipment in the spacecraft with judicious placement of the tracking site.

The cell mounting concepts require development. The design will take advantage of flexible array-printed circuit/wraparound cell contact/induction soldering designs that reduce cell mounting costs while mounting the flexible substrate on inexpensive aluminum sheet to form a panel. The sheet has extruded aluminum stiffeners and provides the mechanical support for the launch dynamic environment.

The solar array wings and drive motor modules are designed such that they can be removed from the spacecraft and replaced for repair or refurbishment operations.

b. <u>Batteries.</u> 15% depth of discharge (DOD) NiCd batteries are employed for operation during eclipse periods and for peak power requirements. A standard battery size (20 amp-hrs) and battery charge controller design is used. The number of battery/battery charge controller sets varies between one and six depending on vehicle load and orbit description.

The standard subsystem supplies unregulated 28 volt dc power. A 28 vdc regulator/converter and a 400 hz, 3-phase, 115 vac inverter will be available as options. Any additional conditioning would be supplied by the using experiment package or spacecraft subsystem. The design of these two units will emphasize a module approach in sizes of about 100 watts that will allow the plugging in of these units on missions requiring them.

Figure 7-17 shows a block diagram of the power supply elements.

c. <u>Battery Charge Controllers.</u> In most spacecraft power systems the battery charging and discharge programming scheme is inextricably tied to the solar array characteristics and the regulation method used to assure a certain

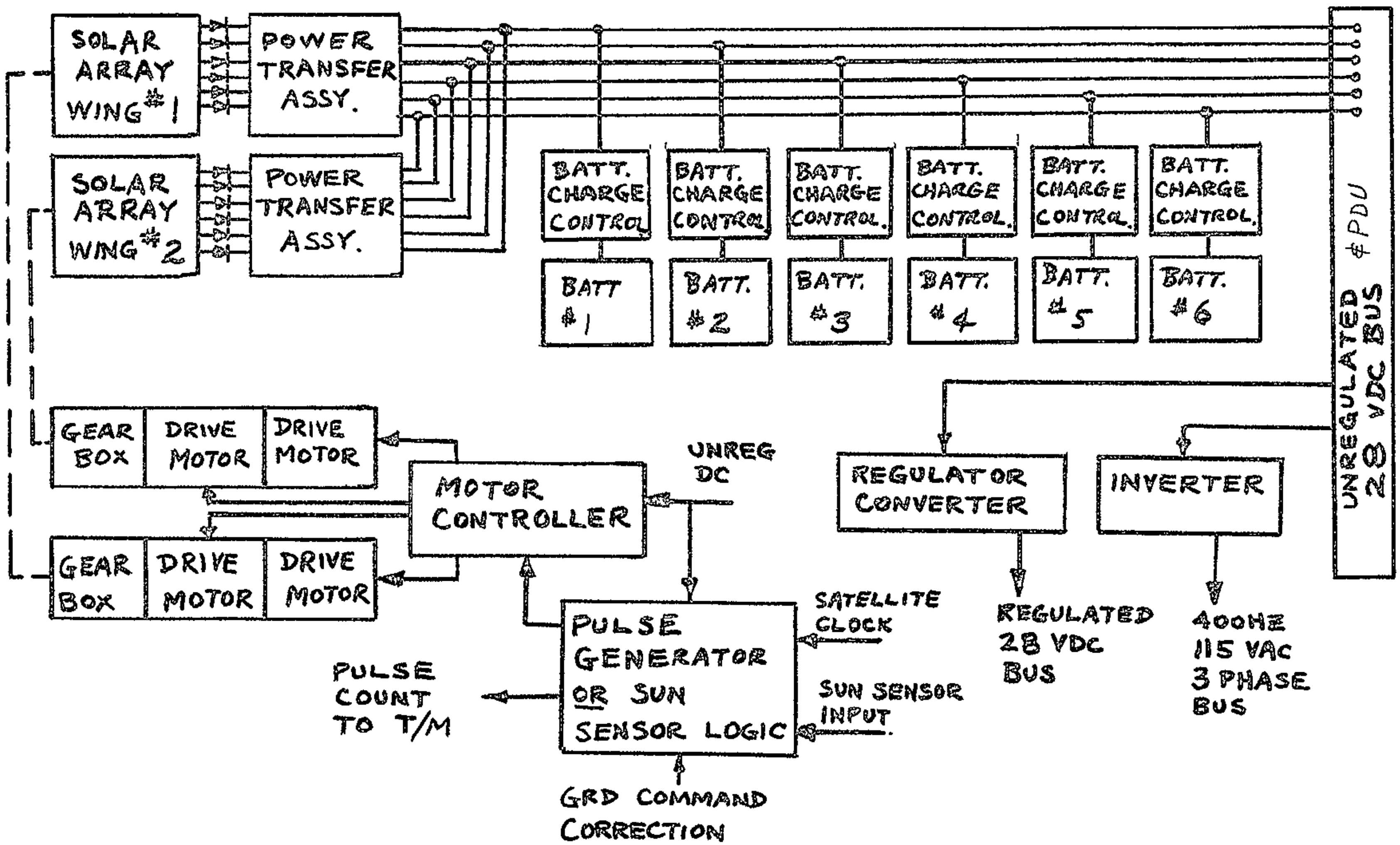


Fig. 7-17 Power Supply Block Diagram, Subsystems A, B & C

operating voltage range of the power system bus.

The standard spacecraft design requirements suggest the following design approach to the battery/power controls system:

(1) Low (15% of rated capacity) battery depth of discharge for long life, simple control requirements, wide temperature operations.

(2) Batteries connected directly to bus (no control intermediate) for low bus impedance, no need for an active discharge control or voltage regulation of solar array.

(3) Charge control via on/off switching of the solar array.

The use of the battery as system "regulator" has been successfully used in several space programs. It is a less expensive approach than separate solar array regulators and charge controllers. Although battery cycling is more frequent, with this system, long life (3 years) has nevertheless been achieved over wide temperature ranges.

Because of the ranges of power levels, and the variation in duty cycles, the control levels of the charge controller would have adjustments corresponding to certain missions designed into the device.

Since the solar array modules are isolated, each feeding individual battery/charge controller inputs, assembly of systems of various power levels for similar missions would amount to addition of identical solar array/charge controller/battery elements.

d. Pyro Programmer. This unit is used in missions where man-controlled operations are not available to provide deployment and other mechanical events.

e. Power Distribution Unit. The unregulated and regulated power buses and associated fuses and current sensors are located in the Power Distribution

Unit. The fuse mounting arrangement, consisting of plug-in fuse modules, offers flexibility whereby fuse ratings, and/or redundancy concepts may be altered as desired.

f. General EPS Characteristics. The characteristics of subsystem options A, B, and C are summarized in Fig. 7-18.

7.4.5.3 EPS Design Approaches - Spin Stabilized Spacecraft.

a. Solar Arrays. The option "D" EPS has body-mounted solar panels either attached to the side panels or deployed on a simple flip-out fixed panel configuration. The selected version of option D employs 8 fixed panels. Solar cells are mounted on one side of the panels and the panels are screw-attached on the sides of the standard spacecraft in eight equally spaced locations around the periphery of the spacecraft

b. Batteries. A pair of 12-ampere-hour batteries are selected for option D based on a 47 percent shade, 53 percent sun orbit with two batteries providing 13 percent DOD.

c. General EPS Characteristics. The option D EPS block diagram is shown on Figure 7-19. Figure 7-20 summarizes the option D characteristics.

7.4.5.4 Special EPS Conditions. The described subsystems are not readily adaptable to power requirements much above 1000 watts; about 7 missions in the model have power requirements in the range from 1.5 to 15 KW. However, although the planetary missions change the distribution of the array area requirements, the variation in the number of array panels which can be added may adequately cover these missions.

If array temperature variations make the single standard solar panel design

	Option A	Option B	Option C
Solar Array Area - ft^2	63.8	127.6	191.4
Nominal Array Temperature - ^{o}C	85	85	85
Average Vehicle Load (B.O.L.)-watts			
$\beta = 0^{o}$ (150 nm alt.)	155	310	465
$\beta = 90^{o}$	370	740	1110
Synchronous Equatorial	350	700	1050
Planetary (1 AU)	370	740	1110
Solar Array Weight - lbs	122	244	366
Solar Array Voltage	0 - 50	0 - 50	0 - 50
Nominal Bus Voltage	23 - 24	23 - 24	23 - 24
Regulated Voltage	28 VDC 115 VAC	28 VDC 115 VAC	28 VDC 115 VAC

Module Number		Size	Wt. (lb.)	Option A	Option B	Option C
1	NiCd Batteries (20 amp-hr)	7x7x18 in.	54.7	2	4	6
	Battery Current Shunt Assy.	4x4x4 in.	1.3	1	1	1
	Charge Controller	8x9x10 in.	6	2	4	6
2	Power Dist. Unit	19x16x5 in	32	1	1	1
	Regulator Convert.	9x8x20	37	1	1	1
	Inverters	9x6x20	35	1	1	1
3	Pulse Generator	6x6x6 in.	6	1	1	1
	Motor Controller	7x8x12 in.	7	1	1	1
	Tracker Motor	4x4x8 in.	9	4	4	4
	Power Transfer Assy.	8x8x12 in.	6	2	2	2
	Solar Array Panels	4.76 ft^2	7.8	12	24	36
1, 2, 3	Power Bus (Cables & Connectors)			1 (107)	1 (214)	1 (320)

$1\ in_2$ - 2.54 cm_2
$1\ ft^2 = 0.093\ m^2$
1 lb = 0.4536 kg

Fig. 7-18 Summary of EPS Options A, B and C

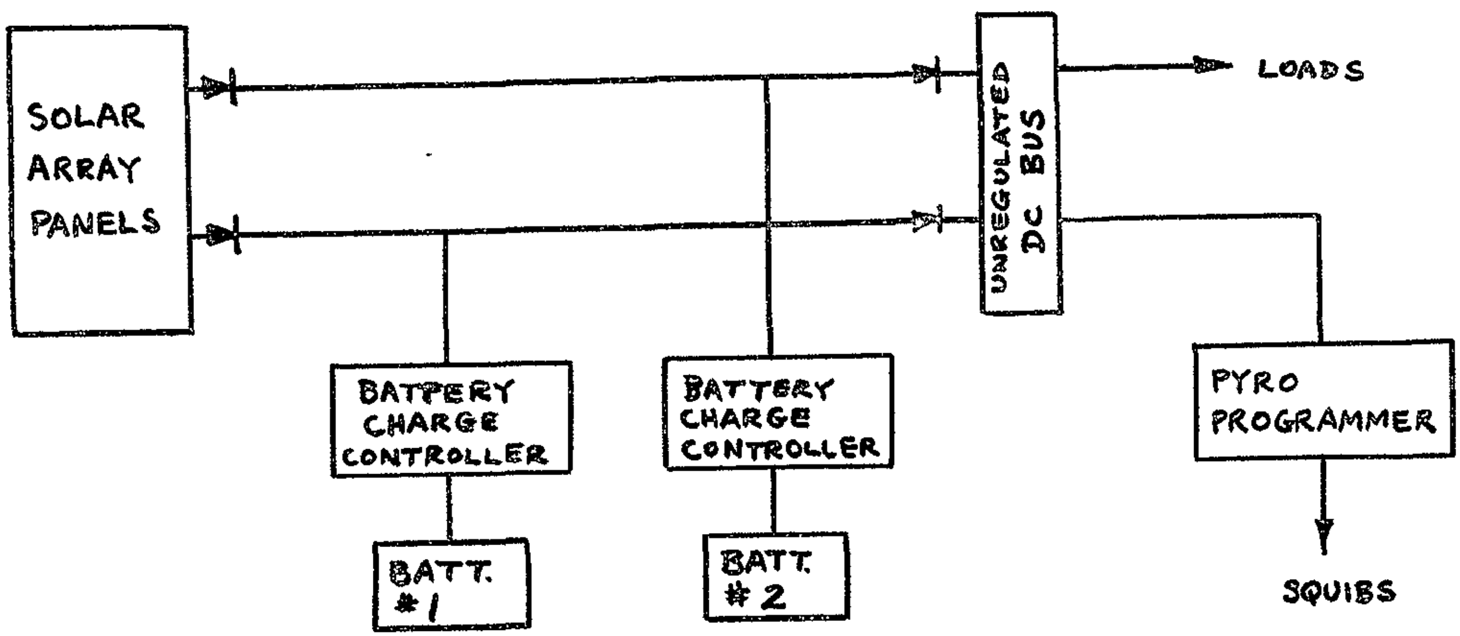


Fig. 7-19 Option D EPS Block Diagram

	Size (in.)	Weight (lb)	Qty.
Solar Array Panel	26.9 x 40	12.5	8
NiCd Batteries (12 AH)	7 x 7 x 10	30.0	2
Battery Charge Controller	6 x 6 x 12	7.0	2
Pyro Programmer	4 x 4 x 5	2.6	1

1 in. = 2.54 cm
1 lb = 0.4536 kg

Fig. 7-20 EPS Equipment List - Option D

unacceptable for some missions, alternatives include:

a. Provide DC/DC converter to change voltage from array to bus requirement

b. Provide more than one standard panel design to split the total temperature range into subranges.

If some missions, e.g., a synchronous orbit, require a more sophisticated battery charge controller design concept than the one described, a variant in the battery charge controller development effort is recommended. Cost savings would be expected by developing and procuring both controllers rather than using only sophisticated design.

7.4.6 Communications - Data Processing and Instrumentation (CDPI) Subsystem

7.4.6.1 CDPI Requirements. The CDPI Subsystem provides the following functions:

a. Tracking: Receive/transmit signals at the spacecraft for position/orbit determination by ground stations within the following limits:

(1) Ranging: 50-ft (15.3m) 1σ) bias error and 60-ft (18.3 m) rms noise error

(2) Range-rate: 0.2 ft/sec (0.061 m/sec)(1σ) 1 sec smoothing

(3) Angle: 1.0 mrad (1σ) bias error, 1.0 mrad rms noise error

b. Telemetry: Provide spacecraft-status data transmission within following limits:

(1) Up to 1 kbps continuous data transmission

(2) Up to 10^5 bps readout of stored data from low earth orbit

(3) Digitized data transmission following on-board or ground computer processing

c. Command: Receive and decode spacecraft commands for:

(1) Digital system - format unspecified

(2) Data rates up to 10 kbps

(3) Stored - program capability for low earth orbit missions

d. Data Handling: Collect, store, process and route data:

(1) Decode, store, route commands

(2) Sample, encode multiplex, store, format telemetry data

- 60-500 channels
- 1/30 to 250 SPS
- A/D conversion up to 8 kits

e. Communications: Transmit experiment data:

(1) Digital data format

(2) 10^2 to 10^7 bps

f. Instrumentation: Sensors to provide spacecraft status signals:

(1) Outputs 0 - 5 volts or 0 - 500 mv full scale

7.4.6.2 CDPI Design Approaches

a. Frequency Selection. The bandwidth capability for the downlink at 2200 to 2300 MHz (S-Band) is adequate for the 10^7 bits maximum data rate. A unified carrier approach is used to minimize the number of RF links needed.

b. Ground Station Parameters. The spacecraft links are designed to operate with relatively inexpensive, and transportable, 30-ft. (9.15m) diameter antennas on the ground. Interplanetary missions will require the 210-ft (64.1m) DSN antennas.

Receiver sensitivity is not critical up to synchronous altitude and system noise temperatures of 250°K can be readily achieved using low-cost, uncooled parametric amplifiers. Interplanetary distances require the use of the DSIF mazer front-end receiver (approximately 25°K).

c. Effective Radiated Power (ERP). The above selected parameters result in the ERP requirements shown in Fig. 7-21. ERP are converted to transmitter power and antenna type in Fig. 7-22.

d. Antennas. Earth-coverage antennas (typically a 20-deg beamwidth) require only modest spacecraft attitude stability (about ± 1.5 deg). The 3 ft diameter dish (10-deg beamwidth) should be steerable but the 10- and 30-ft reflectors (3.06- and 10.2 m) (3 deg and 1 deg beamwidth) require continuous tracking capability.

e. Transmitters. A 0.25 watt power is the basic transmitter output level; higher powers are obtained by addition of power amplifiers; 2.5 watts using solid state and 10 to 50 watts with tube-type amplifiers. The transmitter power-antenna type combinations shown, three options for each regime, satisfy all data transmission requirements of the mission model and introduce little development cost.

f. Data Handling Equipment. The development of low cost spacecraft computers makes it feasible to incorporate a general purpose computer in a majority of the spacecraft for stabilization and control, with additional capacity to perform the data handling functions. Data multiplexing/encoding/formatting and command decoding/programming/verification/distribution can be easily accommodated with the addition of appropriate interface units. Computer software provides the flexibility for accommodating changes in data handling requirements from mission to mission. Limited storage capability for experiment or status data can be provided by the addition of computer memory modules.

Storage and readout of large quantities ($>10^5$ bits) of data is best accomplished with magnetic tape recorders since varying the playback

Operating Regime	Data Rate (bps)	ERP (dbw)
Low earth orbit	10^7	8
	10^5	- 12
Synchronous Orbit	10^7	28
	10^3	- 12
Interplanetary - 1 AU	10^5	52
- 3 AU	10^5	61
- 1 AU	33	17 (omni)
-3 AU	3	17 (omni)

Fig. 7-21 Effective Radiated Power (ERP) Requirements

Operating Regime	Transmitter Power (watts)	Antenna Type
Low Earth Orbit	0.25	Omni
	2.5	Omni
	10.0	Omni
Synchronous Orbit	0.25	Omni & earth coverage
	2.5	Omni & earth coverage
	2.5	3 ft parabolic
Interplanetary	50.0	omni
	50.0	10 ft parabolic
	50.0	30 ft parabolic

1 ft = 0.3048 m

Fig. 7-22 Transmitter Power and Antenna

speed permits matching contact time and link capacity. Although the number of recorder variables precludes selecting a standard unit at this time, a representative unit for low earth orbit missions is a two-track digital recorder with a record-to-playback ratio of 1:26.

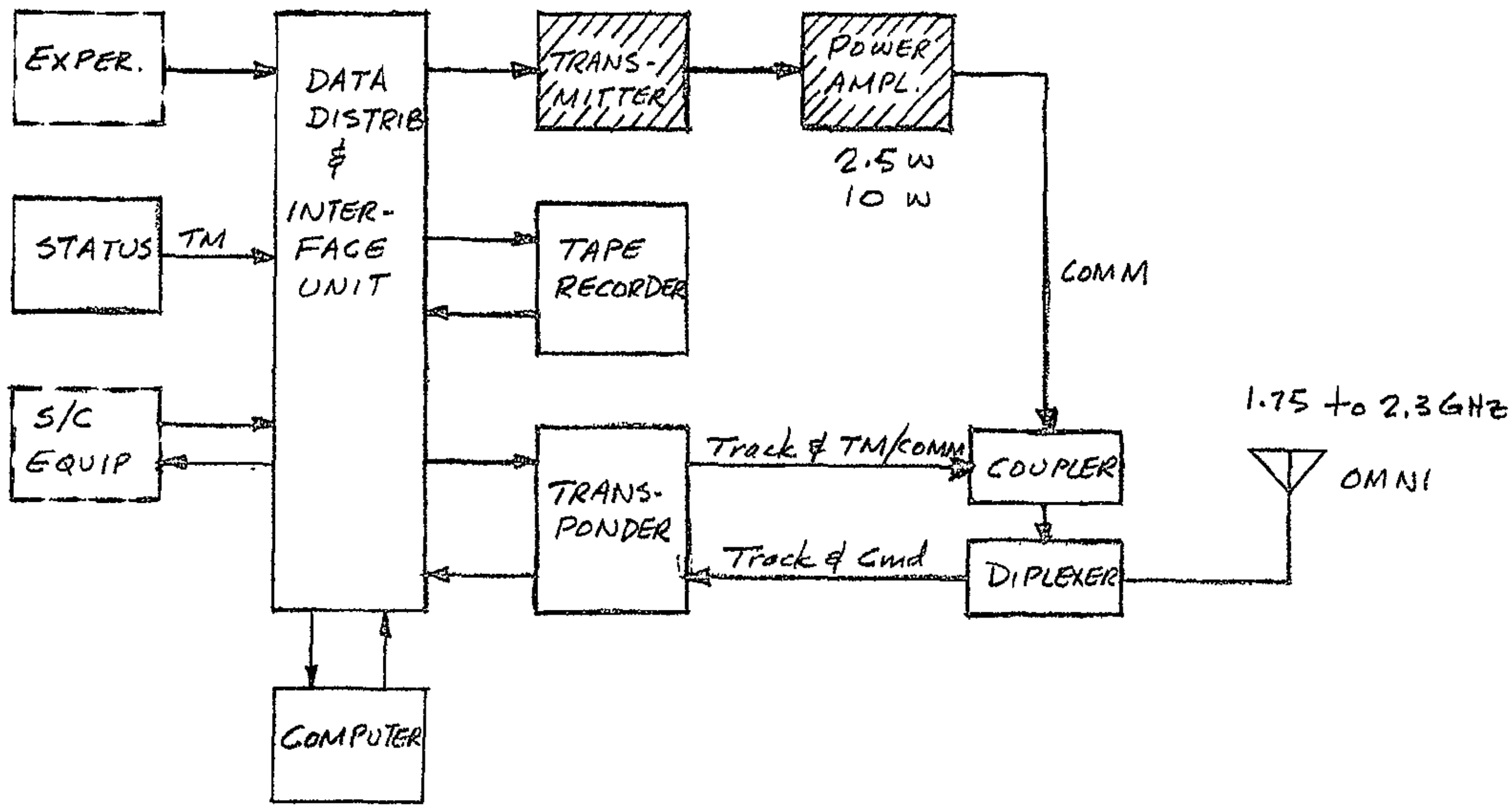
7.4.6.3 CDPI Basic Elements. The Standard CDPI Subsystems are grouped into three categories:

- Low Earth Orbit (LEO): Communication distance ≈ 2200 nm (4080 km)– altitude less than 1000 nm (1852 km)
- Synchronous Earth Orbit (SEO): Communication distance ≈ 22,000 nm (40,800 km)
- Planetary: Communication distance 1 to 3 AU

a. LEO Configuration of CDPI. Figure 7-23 shows the LEO configuration and optional components.

The Data Distribution and Interface Unit (DDIU) provides multiplexing, encoding, formatting, decoding, timing, routing, and processing in conjunction with the computer. This unit is conceived as being flexible enough to handle a variety of data rates and formats to meet all anticipated mission requirements. The computer is expandable in its memory capacity to match the requirements of a particular mission. Multiplex digital experiment and status telemetry data is routed to the tape recorder for storage or directly to the transponder or separate communication transmitter for transmission to the ground stations.

Data rates of 10^6 bps and 10^7 bps require, in addition to power amplifiers of 2.5W and 10W, respectively, a carrier frequency separate from the coherent transponder down-link because of the wide band width requirement. The transponder is of the Unified S-Band (USB) carrier type of NASA's or the USAF's equivalent Space Ground Link Subsystem (SGLS). For discussion purposes, the uplink frequency



OPTION	DATA RATE	XPONDER	XMTR	POWER AMPL
1	3×10^5	0.25 W	0.25W	—
2	4×10^6	—	0.25W	2.5W
3	1.6×10^7	—	0.25W	10 W

optional equipment depending upon data rate

Fig. 7-23 Standard LEO CDPI Subsystem Configuration

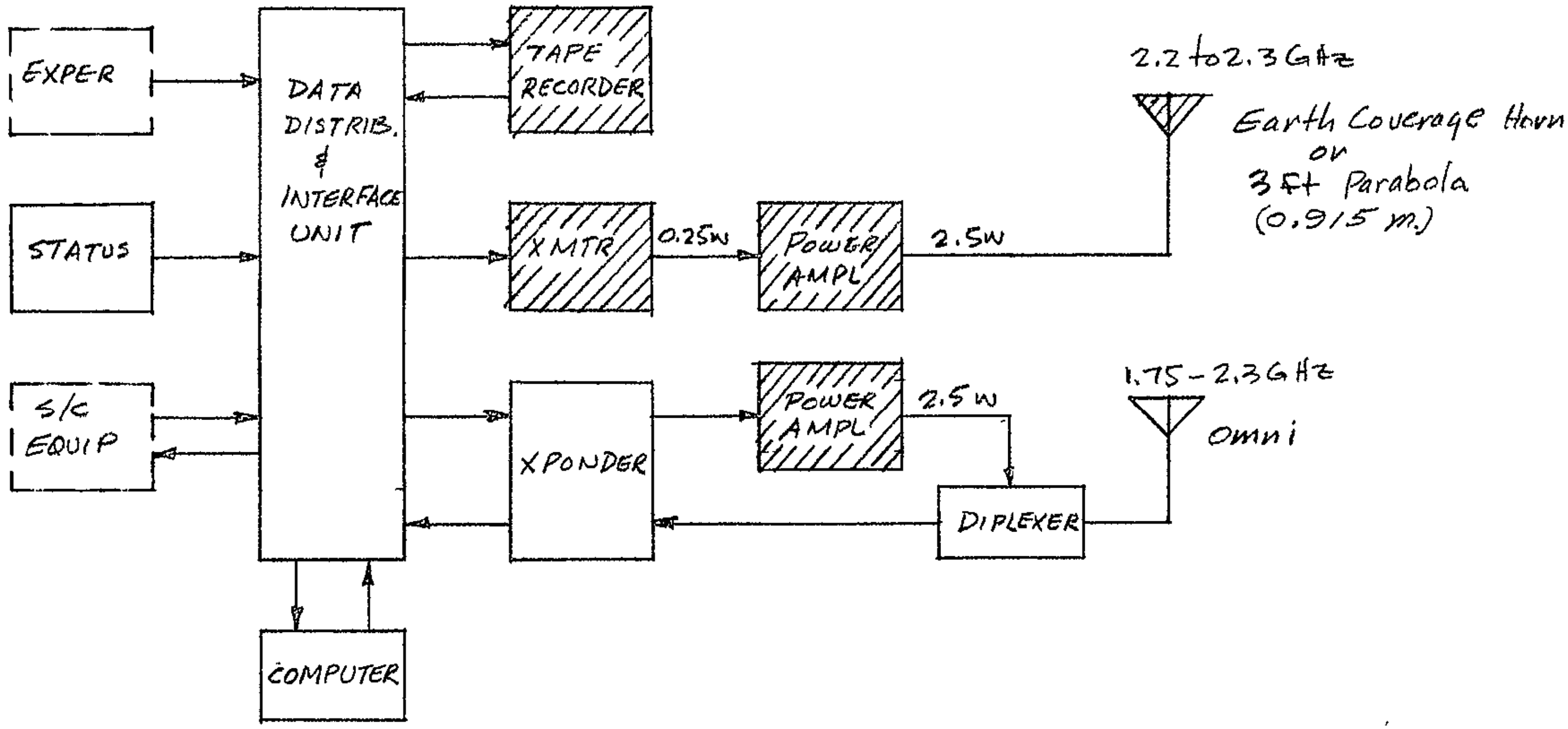
band is assumed to be at 1.75 to 1.85 GHz and the downlink band at 2.2 to 2.3 GHz. The unified carrier transponders receive the uplink carrier modulated with a range-tracking digital code sequence and command subcarriers. The command configuration is demodulated and sent to the DDIU for decoding. The range code is demodulated with telemetry subcarriers onto the down link carrier which is coherently related to the uplink frequency. The coherent carrier allows precise two-way doppler tracking of the spacecraft in addition to range tracking measurements.

The transmit and receive frequencies are isolated from each other by a diplexer to allow operation through a single antenna. The omni-directional antenna permits unrestricted vehicle attitude; a typical antenna would be the "slotted cylinder" type fed by a boom waveguide. Selection of a "standard" omni-type antenna is not possible at this preliminary stage since the pattern is highly dependent upon the spacecraft physical configuration and antenna placement.

b. SEO Configuration of CDPI. The SEO configuration and options are shown in Figure 7-24. The tape recorder would only be required if continuous readout is not possible due to ground station overloading. Some storage capability may also be provided in the computer summary.

c. Interplanetary Configuration for CDPI. The standard CDPI subsystem configuration for interplanetary distances, Figure 7-25, provides all the capability of options shown earlier.

7.4.6.4 CDPI for Spin Stabilized Spacecraft. Spin stabilized spacecraft will not require a general purpose computer for stabilization and control. In such a vehicle it therefore may be less costly and simpler to use separate black boxes to perform the data handling functions. This alternate CDPI subsystem configuration is shown in Figure 7-26. Separate PCM Multiplexer/Encoders are used to gether the experiment and status data and a separate Command Decoder and Command Programmer is used to handle the command information. The remainder of the communication equipment is identical to that of the



OPTION	DATA RATE	XPONDER	XMTR	PA.	ANTENNA
1	3×10^3	0.25W	—	—	omni
2	3×10^4	0.25W	—	2.5	omni
3	10^5	—	0.25	–	E.C.
4	10^6	—	0.25	2.5	E.C.
5	10^7	–	0.25	2.5	3 ft (0.915 m)

▨ optional equipment depending upon data rate

FIG 7-24 STANDARD SEO CDPI SUBSYSTEM CONFIGURATION

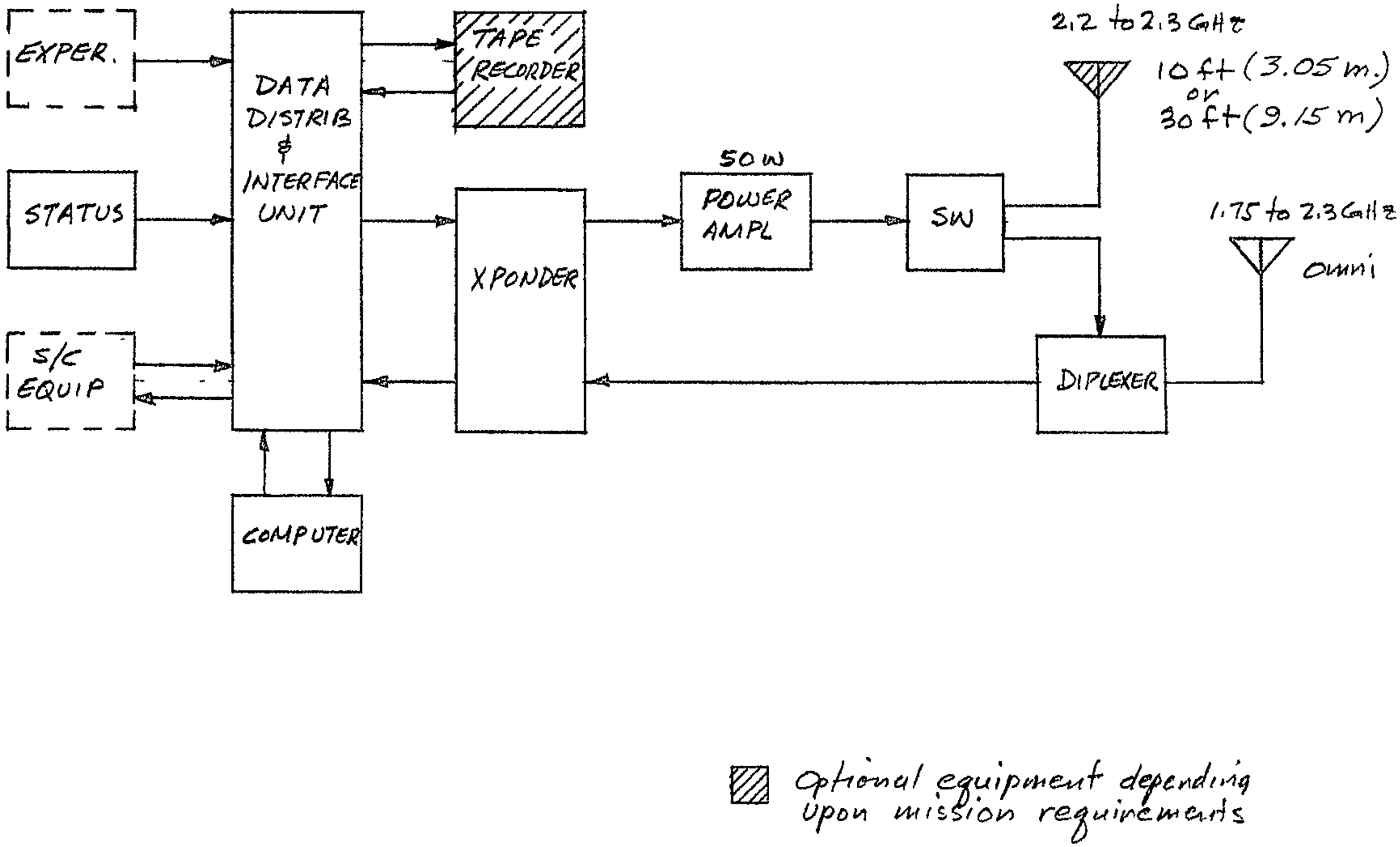


FIG 7-25 STANDARD INTERPLANETARY CDPI SUBSYSTEM CONFIGURATION

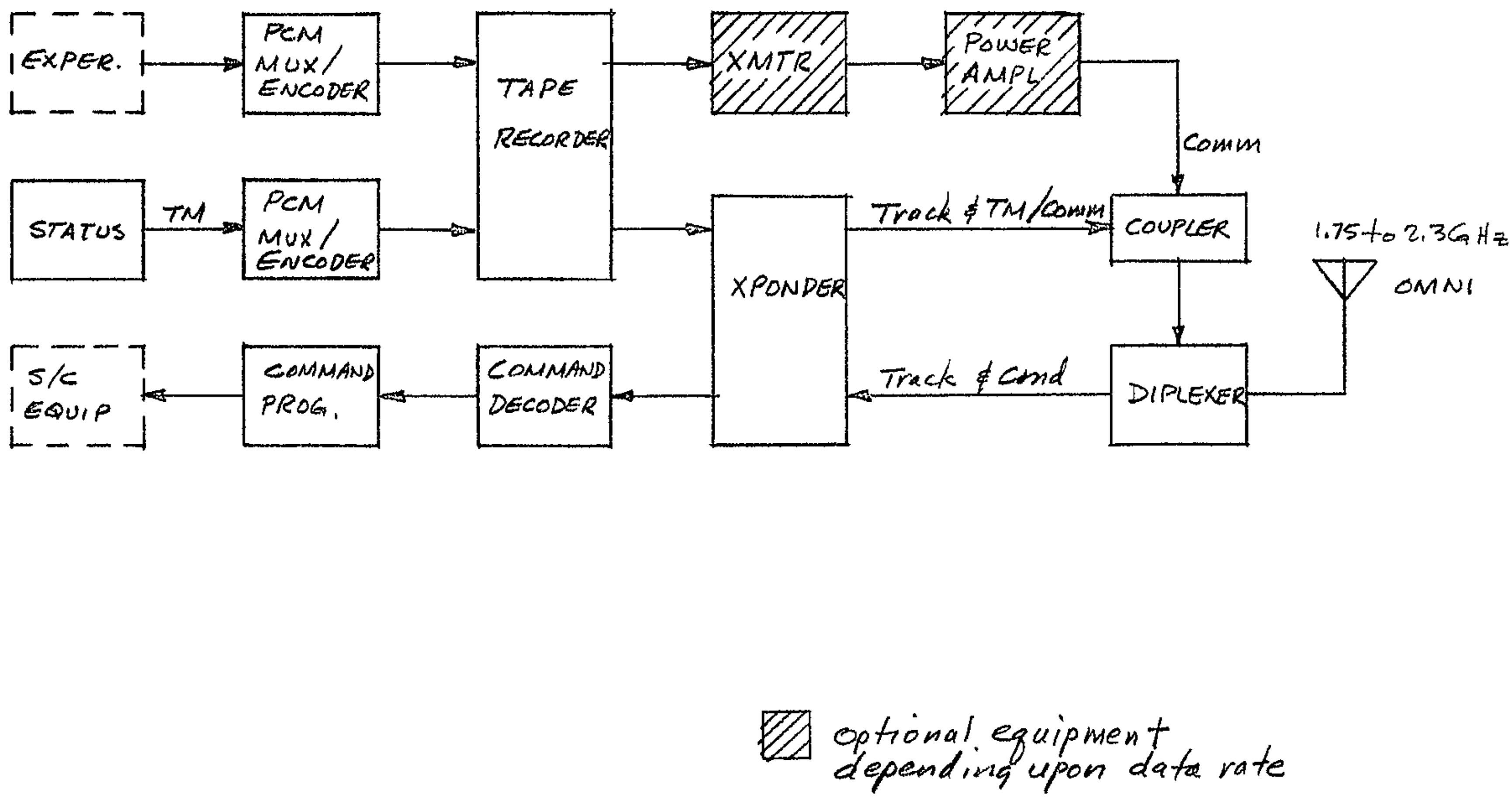


FIG 7-26 ALTERNATE LEO CDPI CONFIGURATION (SPIN-STABILIZED)

previously described CDPI subsystem.

7.4.6.5 Summarization of CDPI Standard Subsystem Options. A summary of the selected options for standardized CDPI subsystems in each of three mission categories is summarized in Figure 7-27. The primary differences between options is the addition of an amplifier and/or different antenna to enable the transmission of higher data rates. The resultant impact on the spacecraft design is a power increase and more stringent antenna/spacecraft pointing requirement.

7.4.6.6 Modularization of CDPI Equipment. The Standard CDPI subsystem is most readily grouped into two basic modules as shown in Figure 7-28. Module 1 contains all of the data handling equipment which is readily removable from the spacecraft. This includes the transponder, transmitter, amplifiers, couplers, and diplexers.

Antennas would normally stay with the spacecraft (non-replaceable). Instrumentation would be distributed throughout the spacecraft in the other subsystems. Estimated maximum size and weights for the two modules are:

	Size	Weight
Data Handling Module	12x24x24 in	60 lbs.
	(30.5x61x61 cm)	(27.2 kg)
RF Module	10x20x20 in	32 lbs.
	(25.4x50.8x50.8 cm)	(14.5 kg)

The size, weight, and power of all the CDPI Subsystem equipment is listed in Figures 7-29a and 7-29b (LEO); Figures 7-30a and 7-30b (SEO); and 7-31 (Interplanetary).

7.4.6.7 Standard CDPI Components. The components were selected on the basis of meeting a maximum number of performance categories with the smallest number of units requiring development, production and inventory.

Mission Category	Option	Data Rate (bps)	Weight (lbs)	Power (watts)	Comments
LEO	1	3×10^5	61.5	73/79	
	2	4×10^6	62	83/89	Add 2.5W amplifier
	3	1.6×10^7	67.5	113/119	Add 10W amplifier
	4	3×10^5	43.5	38/44	Spinner - no computer
SEO	1	3×10^3	77	66	
	2	3×10^4	77.5	76	Add amplifier
	3	10^5	82	69	Add transmitter and and E.C. Antenna *
	4	10^6	82.5	79	Add transmitter and E. C. Antenna *
	5	10^7	82.5	79	Add transmitter, amplifier, and 3-foot antenna; antenna steering required
Interplanetary	1	10^5 at 1 AU	93.3	271	10-foot tracking antenna
	2	10^5 at 3 AU	273	271	30-foot tracking antenna

*E. C. = earth-coverage horn

1 lb = 0.4536 kg

1 ft = 0.3048 m

Fig. 7-27 Standard CDPI Subsystem Summary

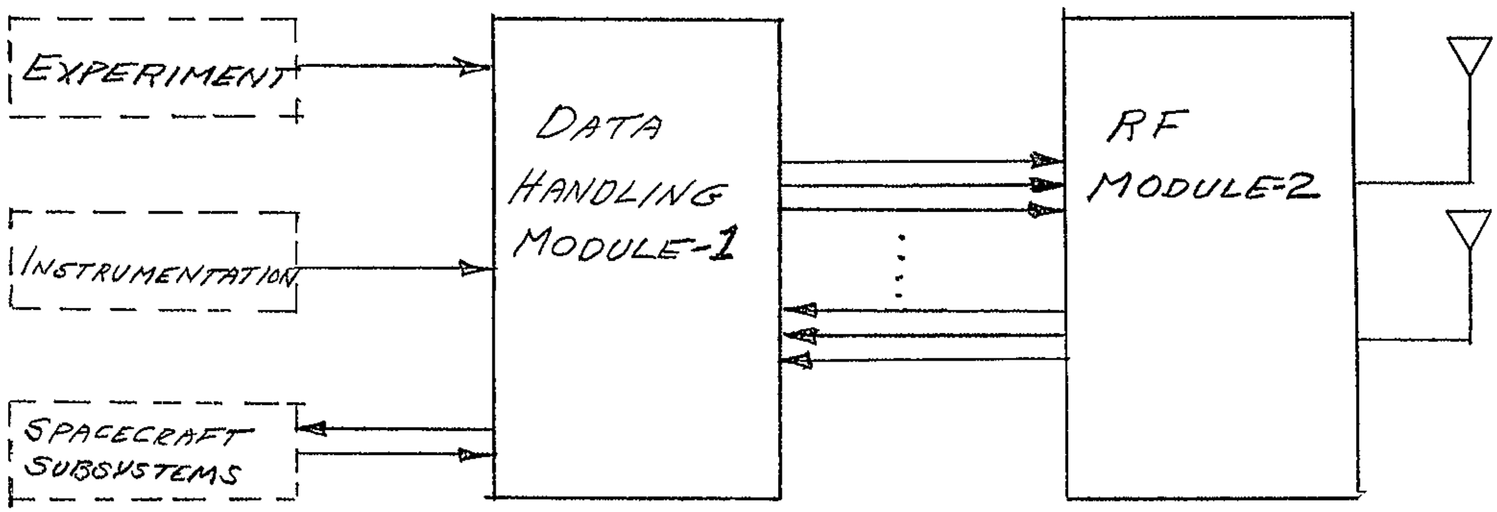


FIGURE 7-28

CDPI MODULES AND INTERCONNECTIONS

Component	Size, in (cm)	Weight, lbs (kg)	Power (watts)
Option 1: Data rates $\leqq 3 \times 10^5$ bps			
Data Dist. and Interface Unit	9 x 12 x 12 (22.9 x 30.5 x 30.5)	35 (15.9)	30
Computer	8 x 8 x 10 (20.3 x 20.3 x 25.4)	16 (7.3)	20
Transponder	16 x 8 x 6 (40.6 x 20.3 x 15.2)	12 (5.5)	16
Transmitter	2 x 5 x 6 (5.1 x 12.7 x 15.2)	2 (0.91)	3
Diplexer	2 x 5 x 4 (5.1 x 12.7 x 10.3)	2 (0.91)	-
Hybrid Coupler	0.5 x 3 x 3 (1.27 x 7.6 x 7.6)	0.5 (0.23)	-
Omni Antenna	8 dia x 16 L (20.3 dia x 40.6 L)	2 (0.91)	-
Tape Recorder	7 x 8 x 9 (17.8 x 20.3 x 22.9)	8 (3.6)	4/10
Totals		77.5 lbs (35.2 kg)	73/79 W
Option 2: Data rates $\leqq 4 \times 10^6$			
Same as Option 1 except add:			
Power Amplifier, 2.5W	2 x 2 x 3 (5.1 x 5.1 x 7.6)	0.5 (0.23)	10
Totals		78 lbs (35.5 kg)	83/89 W
Option 3: Data rates $\leqq 1.6 \times 10^7$			
Same as Option 1 except add:			
Power Amplifier, 10W	3 x 6 x 12 (7.6 x 15.2 x 30.4)	6 (2.7)	40
Totals		83.5 lbs (38 kg)	123/129

Fig. 7-29a LEO CDPI Summary (Sheet 1 of 2)

Component	Size, in (cm)	Weight, lbs (kg)	Power (watts)
Option 4: Data rates $\leqq 3 \times 10^5$; Spinner			
PCM MUX/Encoder (1024 BPS, 128 CH)	8 x 6 x 6 (20.3 x 15.2 x 15.2)	5 (2.27)	5
PCM MUX/Encoder (1024 BPS, 128 CH)	8 x 6 x 6 (20.3 x 15.2 x 15.2)	5 (2.27)	5
Command Decoder (128 COMNDS)	9 x 6 x 6 (22.9 x 15.2 x 15.2)	6 (2.73)	2
Command Programmer (16-Event, Variable Delay)	6 x 5 x 8 (15.2 x 12.7 x 20.3)	7 (3.18)	3
Tape Recorder (120 min., 2-track)	7 x 8 x 9 (17.8 x 20.3 x 22.9)	8 (3.63)	4/10
Transponder	16 x 8 x 6 (40.7 x 20.3 x 15.2)	12 (5.45)	16
Transmitter	2 x 5 x 6 (5.1 x 12.7 x 15.2)	2 (0.91)	3
Diplexer (100 db isolation)	2 x 5 x 4 (5.1 x 12.7 x 10.3)	2 (0.91)	-
Hybrid Coupler	0.5 x 3 x 3 (1.3 x 7.6 x 7.6)	0.5 (0.27)	-
Omni Antenna	8 dia x 16 L (20.3 dia x 40.6)	2 (0.91)	-
Totals		49.5 lbs (22.5 kg)	38/44

Fig. 7-29b LEO CDPI Summary (Sheet 2 of 2)

Component	Size, in (cm)	Weight, lbs (kg)	Power (watts)
Option 1: Data rates $\leqq 3 \times 10^3$ bps			
DDIU	9 x 12 x 12 (22.9 x 30.5 x 30.5)	35 (15.9)	30
Computer	8 x 8 x 10 (20.3 x 20.3 x 25.4)	16 (7.3)	20
Transponder	16 x 8 x 6 (40.7 x 20.3 x 15.2)	12 (5.5)	16
Omni Antenna	8 dia x 16 L (20.3 dia x 40.6 L)	2 (0.91)	-
Diplexer	2 x 5 x 4 (5.1 x 12.7 x 10.3))	2 (0.91)	-
Totals		67 lbs (30.5 kg)	66 W
Option 2: Data rates $\leqq 3 \times 10^4$ bps			
Same as Option 1 except add:			
Power Amplifier, 2.5W	2 x 2 x 3 (5.1.x 5.1 x 7.6)	0.5 (0.22)	10
Total		67.5 lbs (30.7 kg)	76 W
Option 3: Data rates $\leqq 10^5$			
Same as Option 1 except add:			
Transmitter	2 x 5 x 6 (5.1 x 12.7 x 15.2)	2 (0.91)	3
E.C. Horn Antenna	18 dia x 24 L (45.7 dia x 61 L	3 (1.36)	-
Totals		72 lbs (32.7 kg)	.69 W

Fig. 7-30a SEO CDPI Summary (Sheet 1 of 2)

Component	Size, in (cm)	Weight, lbs (kg)	Power (watts)
Option 4: Data rates $\leq 10^6$			
Same as Option 3 except add:			
Power Amplifier, 2.5W	2 x 2 x 3 (5.1 x 5.1 x 7.6)	0.5 (0.23)	10
Total		72.5 lbs (33.0 kg)	79 W
Option 5: Data rates $\leq 10^7$			
Same as Option 1 except add:			
Transmitter	2 x 5 x 6 (5.1 x 12.7 x 15.2)	2 (0.91)	3
Power Amplifier, 2.5W	2 x 2 x 3 (5.1 x 5.1 x 7.6)	0.5 (0.23)	10
3-foot (0.92m) Antenna	3-foot dia (0.92-m dia)	3 (1.36)	-
Totals		72.5 lbs (33.0 kg)	79 W

Fig. 7-30b SEO CDPI Summary (Sheet 2 of 2)

Component	Size, in (cm)	Weight, lbs (kg)	Power (watts)
Option 1: 10^5 bps at ≦1 AU			
DDIU	9 x 12 x 12 (22.9 x 30.5 x 30.5)	35 (15.9)	30
Computer	8 x 8 x 10 (20.3 x 20.3 x 25.4)	16 (7.28)	20
Transponder	16 x 8 x 6 (40.6 x 20.3 x 15.2)	12 (5.5)	16
Power Amplifier, 50W	3 x 6 x 12 (7.6 x 15.3 x 30.5)	6 (2.72)	200
RF Switch	1.5 x 1.5 x 3 (3.8 x 3.8 x 7.6)	0.3 (0.13)	-
Diplexer	2 x 5 x 4 (5.1 x 12.7 x 10.2)	2 (0.91)	-
Omni Antenna	8 dia. x 16L (20.3 x 40.6)	2 (0.91)	-
10-foot (3.05m) Antenna	10-foot dia (3.05-m dia)	20 (9.75)	5
Totals		93.3 lbs (42.4 kg)	271 W
Option 2: 10^5 bps at ≦3 AU			
Same as Option 1 except replace 10-foot (3.05m) Antenna with 30-foot (9.15m) Antenna:			
30-foot (9.15-m) Antenna	30-foot dia (9.15-m dia)	200 (91.5)	5
Totals		273 lbs (124 kg)	271 W

Fig. 7-31 Interplanetary CDPI Summary

a. Data Distribution and Interface Unit. This unit must be designed and developed to operate with a designated general-purpose computer. Its design requirements can be determined from a detailed analysis of the mission model. Preliminary specifications for the unit are:

Inputs:	Multiplexer:	64 to 512 channels
	Command:	1 to 10 kbps
	Computer:	TBD
Outputs:	Experiment data:	10^3 to 10^7 bps PCM
	Telemetry data:	10^3 to 10^5 bps PCM
	Command:	32 to 256 commands

Functions:
- Multiplex
- Analog to digital conversion, 8-bit
- Verify and decode commands
- Route data
- Provide system timing

b. Computer. This unit is described as part of the standard S&C subsystem.

c. Transponder. This is a USB-type (or SGLS) transponder; it is common to all Standard CDPI Subsystems. Two models are required because of the different receiver tracking bandwidths and range code requirements; one for all missions at or below synchronous altitudes and the second for interplanetary-type missions.

Preliminary specifications indicate a receiver dynamic range of -40 to -12.7 dbm is desirable.

d. Power Amplifier. Typical specification for the power amplifiers are:

	Power Output		
	2.5 Watt	10 Watt	50 Watt
Frequency (GHz)	2.2 - 2.3	2.2 - 2.3	2.2 - 2.3
Bandwidth (MHz)	5	100	100
Gain (db)	10	13	23

e. Antenna. Typical antenna specifications are:

Type	Frequency (GHz)	Gain (db)	Remarks
Omni	1.75 - 2.3	0	Slotted cylinder
Earth Coverage	"	+15	20 deg BW
3-foot (0.91m)	"	+24	Limited steering required
10-foot (3.05m)	1.75 - 2.2	+34	Monopulse receive feed furlable
30-foot (9.15m)	1.75 - 2.3	+44	Furlable

7.4.7 Attitude Control and Station-Keeping, Propulsion Subsystem (ACS)

7.4.7.1 ACS Requirements. Spacecraft propulsion requirements for attitude control, station keeping, and, where applicable, planetary orbiter injection, were extracted from NASA Mission Model projections or were estimated where these lacked definition. Attitude control and station keeping requirements determine the attitude control system (ACS), whereas planetary orbiter injection requirements will size the main propulsion system.

a. Total Impulse. Attitude control requirements are configuration-dependent. It was therefore necessary to develop scaling relationships for using the low-cost SEO as a reference. Applying these relationships to the programs in the Mission Model produces the distribution of attitude control total impulses shown in Figure 7-32. For station-keeping requirements, the low cost SEO was again used as reference. The total impulse requirements of SEO are as follows:

3750 lb-sec (16680 newton-sec) for Station Keeping
500 lb-sec (2220 newton-sec) for Attitude Control
750 lb-sec (3335 newton-sec) Contingency
______
5000 lb-sec (22200 newton-sec) Total

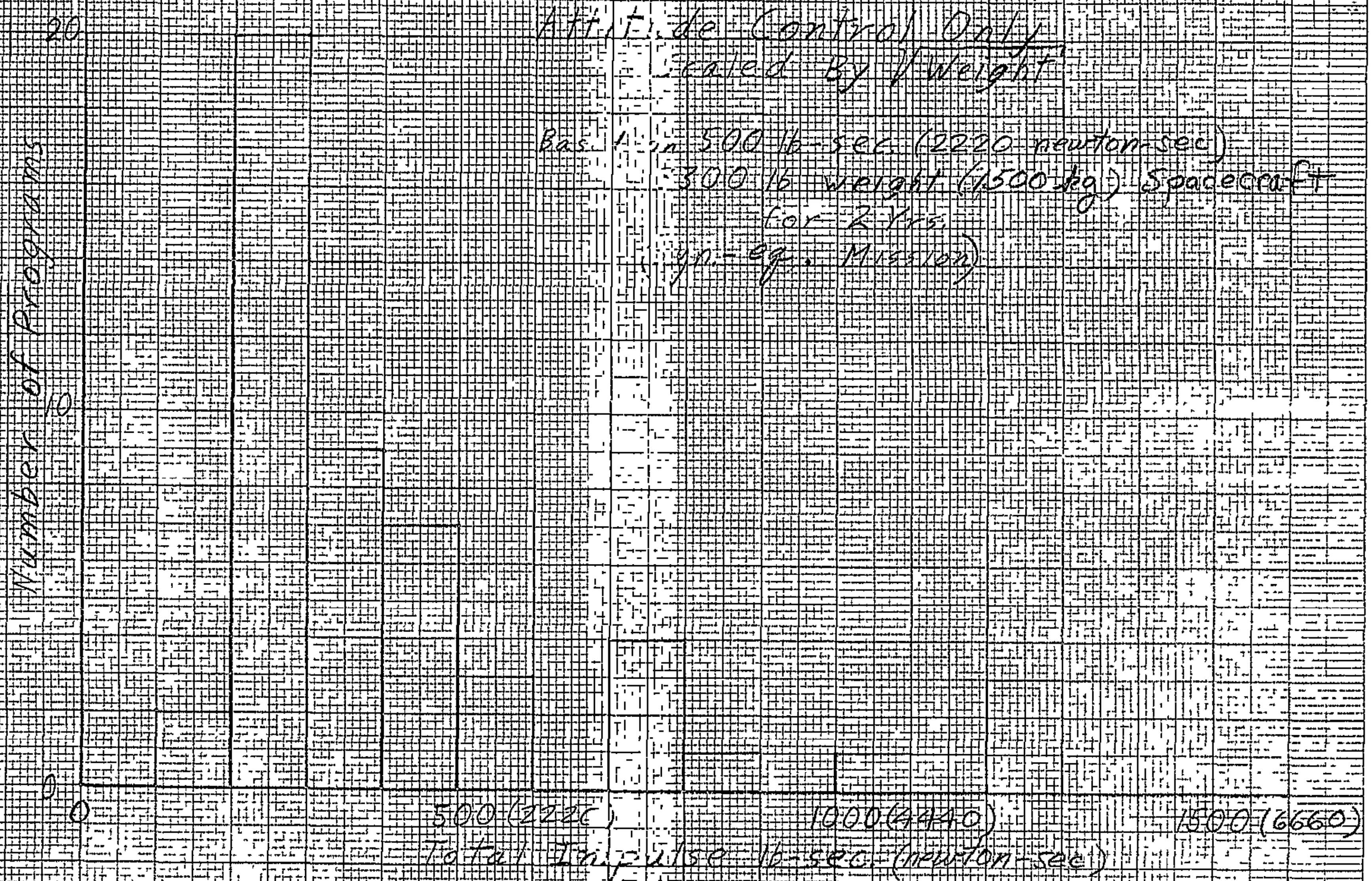

Attit de Control Only
Scaled By √Weight
Bas on 500 lb-sec (2220 newton-sec)
300 lb weight (1500 kg) Spacecraft
for 2 Yrs
yn-eq. Mission)
Number of Programs
20
10
0
0
500 (2220)
1000 (4440)
1500 (6660)
Total Impulse lb-sec (newton-sec)

Figure 7-32

To determine the total impulse for the other missions, this impulse was scaled by weight directly assuming that the stationkeeping required of a synchronous equatorial satellite is representative of the mission model. The resulting computed impulse requirements are plotted in Figure 7-33. Thus, depending upon whether or not there is a translation requirement, there is a concentration of impulse into one of two ranges: (1) 200 to 500 lb-sec (890 to 2,220 newton-sec) or (2) 1,000 to 4,000 lb-sec (4,450 to 17,800 newton-sec). Heavier payloads in the program Mission Model which must perform a "stationkeeping" operation, would require total impulse in the 20,000 to 30,000 lb-sec (89,000 to 133,500 newton-sec) range.

b. Thrust Level. The thrust levels must be selected to have a small enough pulse width to be capable of maintaining the vehicle within the proper control band and also be large enough to perform the following functions: (1) unload a momentum wheel, (2) slew the vehicle to a new position, and (3) position vehicle for docking. Initially, four thrust levels: 0.2, 0.5, 3.5, and 5 lb were considered to cover the spectrum of applications. Later this was reduced to two: 0.5 and 5.0 lbs (2.22 and 22.2 newtons).

c. Propellant Selection. The type of ACS propulsion system to be used has been selected with heavy weighting in the direction of technology status and reliability. According to past experience the smallest ACS requirements, 200 to 500 lb-sec (890 to 2,220 newton-sec) total impulse and 0.2 lb (0.89 newton) thrust most logically would be satisfied by cold gas. The second and third groups, with 1,000 to 30,000 lb-sec (4,440 to 133,500 newton-sec) total impulse and 0.5 to 5.0 lb (2.2 to 22.2 newton) thrust, fall into monopropellant hydrazine regime. However, with pulsing, hydrazine systems can produce both the high and low thrust levels whereas with cold gas systems (N_2/CF_4) storage volumes become unwieldy for the projected upper total impulse levels of safe operating pressure, 3000 psia (2070 newton per cm^2). In the interest of commonality hydrazine has been

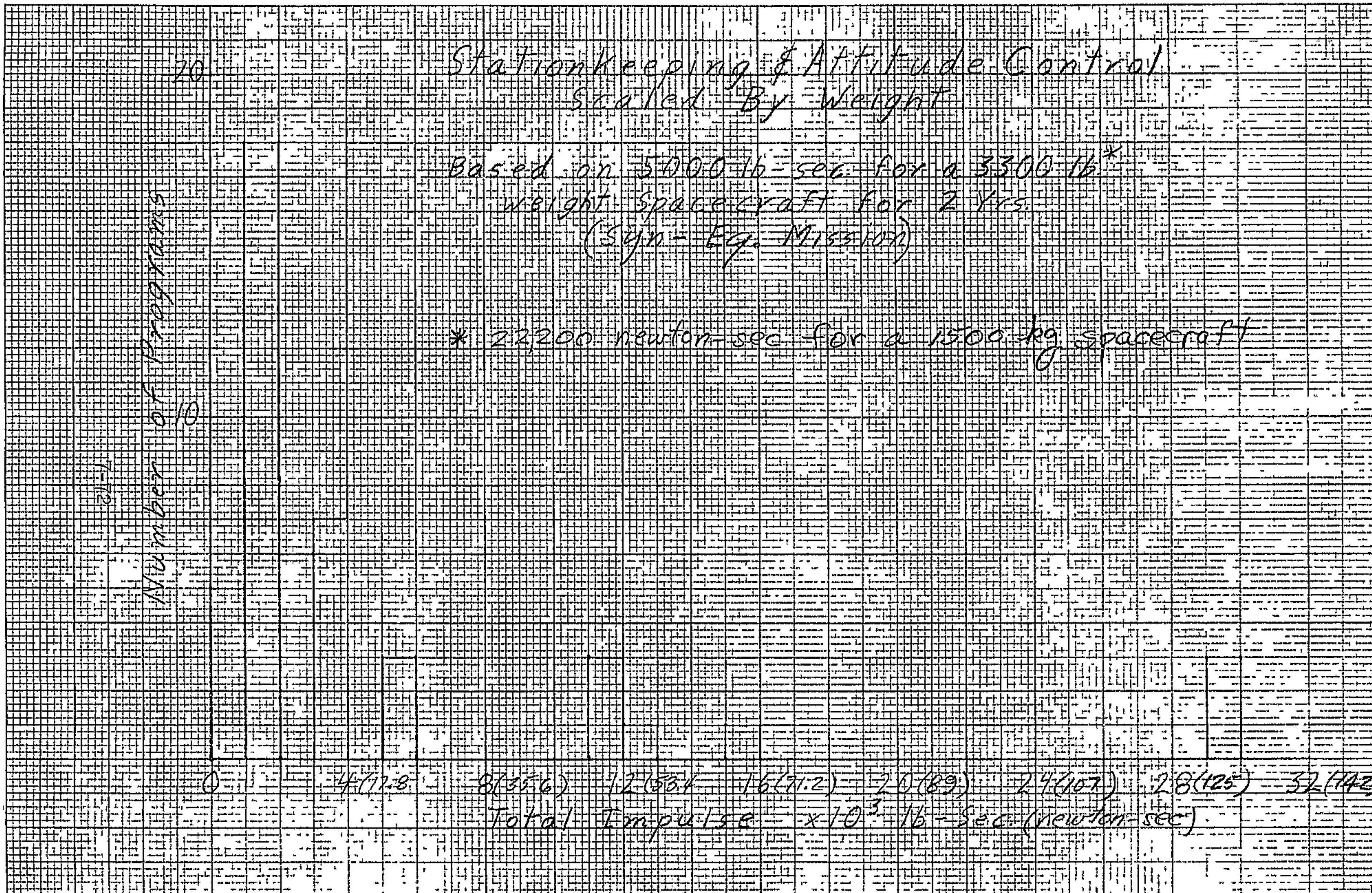

Stationkeeping & Attitude Control
Scaled By Weight
Based on 5000 lb-sec for a 3300 lb* weight Spacecraft for 2 Yrs.
(Syn-Eq. Mission)
* 22,200 newton-sec for a 1500 kg spacecraft
Number of Programs
20
10
0
4(17.8)
8(35.6)
12(53.4)
16(71.2)
20(89)
24(107)
28(125)
32(142)
Total Impulse x10³ lb-Sec (newton-sec)

Figure 7-33

selected to fulfill both requirements on the standard spacecraft. Monopropellant hydrazine systems offer the advantages of relatively high performance with inherent simplicity. Hydrazine systems in this thrust range have demonstrated a steady-state specific impulse of 200 sec. The simplicity of the hydrazine design comes from the use of only one propellant, from the modest decomposition temperature and from the use of a blow-down pressurization system. A reliability of better than 0.998 can be projected.

7.4.7.2 Hydrazine ACS Description. The Attitude Control Subsystem consists of four identical modules, one of which is schematically shown in Figure 7-34. Using monopropellant hydrazine with Shell 405 catalyst, the subsystem operates in a blowdown mode with nitrogen as the pressurant gas. Surface tension devices, as shown in Figure 7-35, are used for controlling the propellant orientation in the tank. This passive propellant management device ensures delivery of gas-free propellant to the thruster assembly under any condition of operation.

The subsystem employs both active and passive thermal controls to prevent propellant freezing and to reduce ignition delay and pressure spiking during low-duty-cycle pulse-mode operations.

The redundancy provided in the module includes dual-series propellant valves, and redundant heaters. Redundancy is not considered necessary for the other components in the system because of their passive operation in flight and demonstrated high reliability. Two standard ACS modules were sized as described below and summarized in Figure 7-36. The small standard module was designed so that three modules could supply 6000 lb-sec (26,700 newton-sec) total impulse. This amount will accommodate the majority of the using program. Each spacecraft, however, is designed to carry four standard modules or 8,000 lb-sec (35,600 Newton-sec) total impulse. The additional impulse is useful in the event one or more propellant valves fail. The large standard module was designed to the same ground rules except that the maximum total impulse

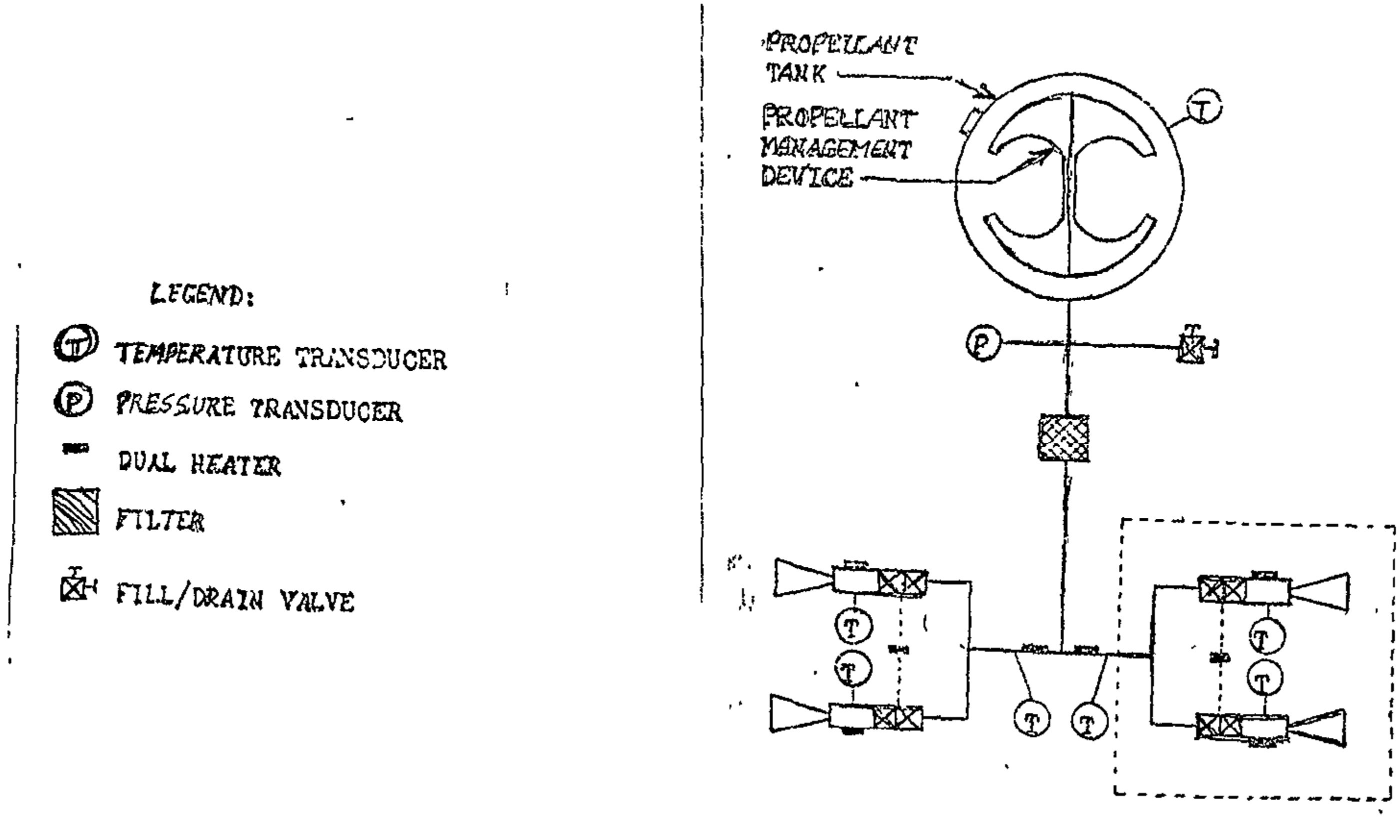


Fig. 7-34 Propulsion System Schematic - Attitude Control Module

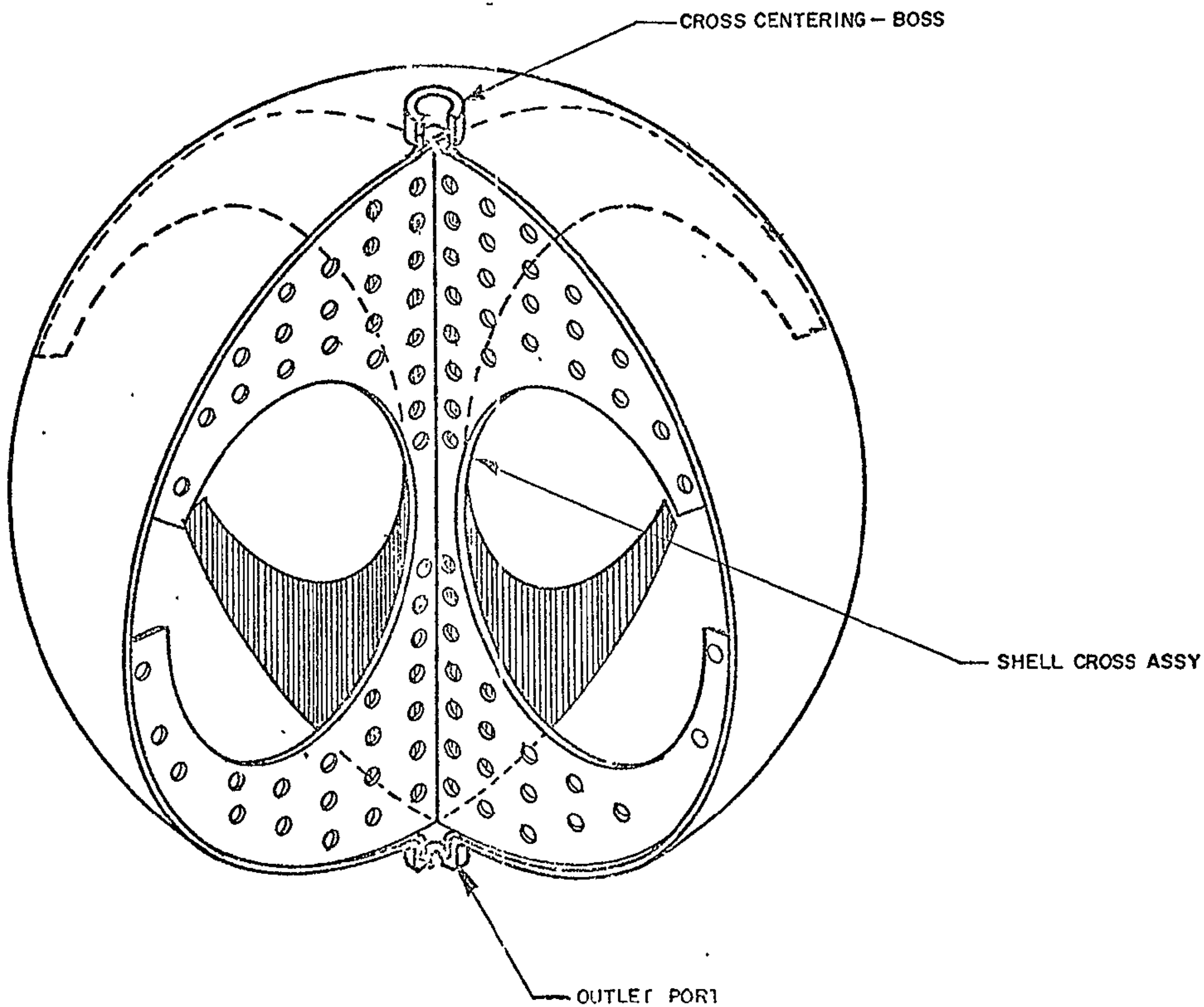


Fig. 7-35 Symmetrical Cross Acquisition Device - Propellant Tank Assembly

ITEM	DESCRIPTION	WEIGHT	
		SMALL MODULE	LARGE MODULE
Thruster Ass'y. (1)	Cluster of four nozzles: 0.5 lbf (2.22 newton) (small); 5.0 lbf 22.2 newton) (large)	6.0 lb	9.0 lb
Hydrazine Tank (1)	12.5" /21" (31.7/53.3 cm) ID titanium, 450 psia (310 newton-cm^2) maximum	7.0	15.0
Fill/Drain Valve (1)	Manually-operated, double seals	0.2	0.2
Pressure Transducers (1)	0-500 psia (0 to 345 newton-cm^2)	0.5	0.5
Temperature Transducers (6)	Two @ 0-2000°F (273 - 1320°K) Two @ 0-300°F (273 - 423°K)	0.7	0.7
Filter (1)	10 micron (10x10^{-3}mm) (norm) 25 micron (25x10^{-3}) (abs)	0.5	0.5
Heaters (18)	All but four thermostatically controlled	2.0	2.0
Plumbing, Fittings, etc.	Stainless	1.5	1.5
Hydrazine Propellant		11	53

1 lb = 0.4536 kg

Figure 7-36 Equipment List - ACS Modules

is 30,000 lb-sec (133,400 newton-sec) from four modules.

With the hydrazine system operating in a blowdown mode, thrust varies with the feed pressure. Therefore by changing the propellant loading both total impulse and thrust can be varied. For the two thruster sizes, nominally 0.5 and 5.0 lbf (2.2 and 22.2 newton), Figure 7-37 shows the relationship of thrust and feed pressure.

7.4.7.3 Main Propulsion Requirements. Some spacecraft missions involve major trajectory changes such as orbital plane change or injection into circum-planetary orbit. In order to keep burn times within reasonable limits the associated thrust levels have to be higher than offered by the ACS. The question arises whether such main propulsion requirements are sufficiently recurring to be candidates for standardization. A survey of the NASA Mission Model shows that only planetary missions require a main propulsion system in the spacecraft. Earth orbital mission uses the last launch vehicle stage (upper stage) for such maneuvers. It was found that maneuver requirements of spacecraft going beyond Mars exceed the capabilities of what could reasonably be called spacecraft propulsion, and they are therefore not considered here. Also, planetary landers and payload return missions were excluded because they have propulsion requirements that would not be amenable to standardization.

Main propulsion requirements were determined from orbit injection velocity requirements assuming a thrust/weight ratio of about 1/10. Figure 7-38 shows the main propulsion requirements of five programs containing eight flights. Although some modularity in requirements can be seen, the incidence is too small to make standardization worthwhile.

7.4.8 Stabilization and Control (S&C) Subsystem

7.4.8.1 S&C Subsystem Requirements. The Standard Spacecraft Stabilization and Control Subsystem has the following functions:

- Stabilize the spacecraft following shuttle separation, and establish

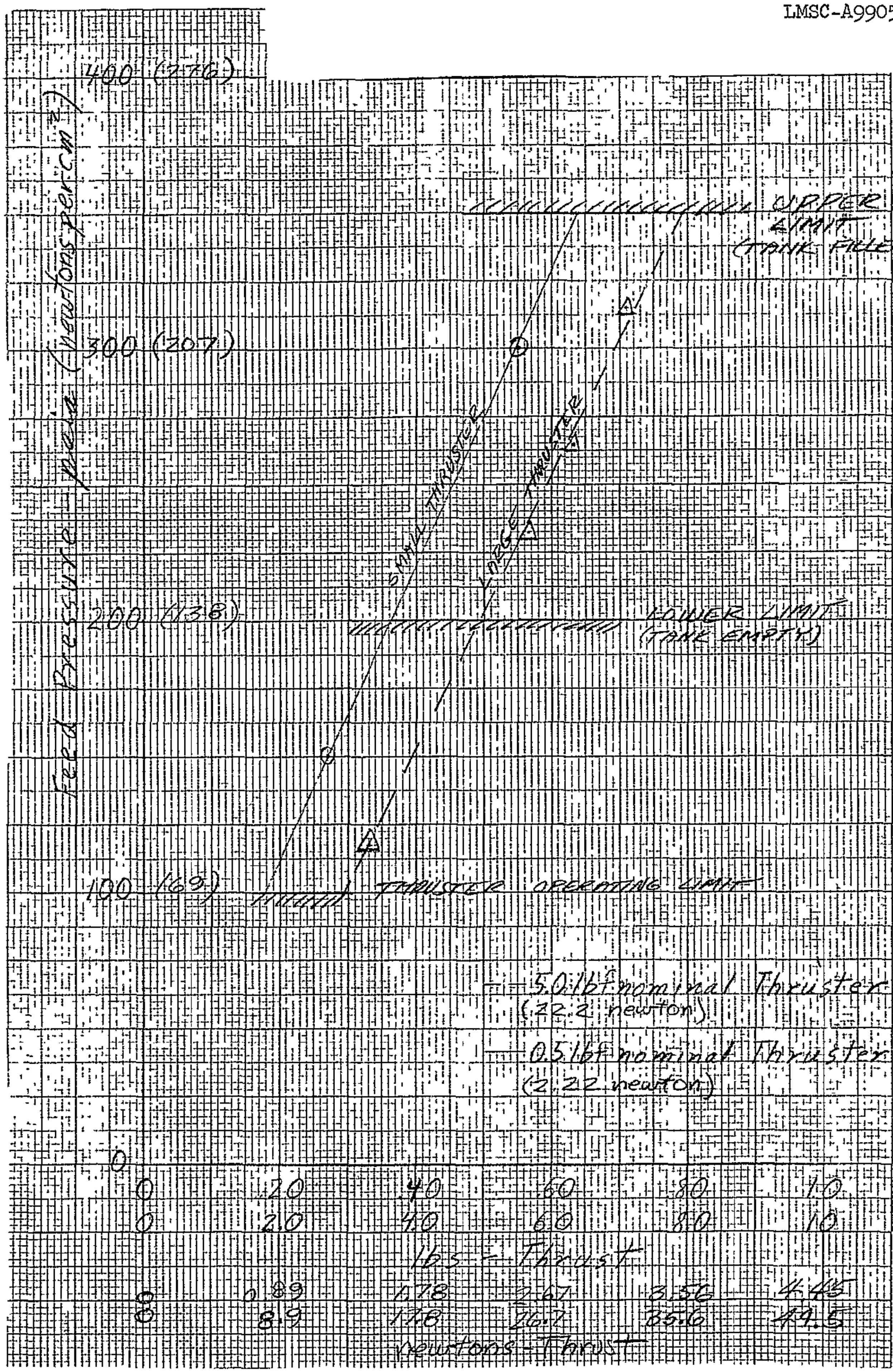


Fig. 7-37 Hydrazine Blowdown Operation

	Programs	Flights	Thrust
	3	5	@ T ≈ 800 lbf (3560 Newtons)
	2	3	@ T ≈ 100 lbf (445 Newtons)
Totals	5	8	

	Programs	Flights	Total Impulse
	2	3	@ I_T ≈ 1,530,000 lb sec (6,800,000 Newton-sec)
	1	2	@ 700,000 lb-sec (3,111,000 Newton-sec)
	2	3	@ 200,000 lb-sec (888,000 Newton-sec)
Totals	5	8	

Fig. 7-38 Orbit Injection Requirements of Planetary Spacecraft

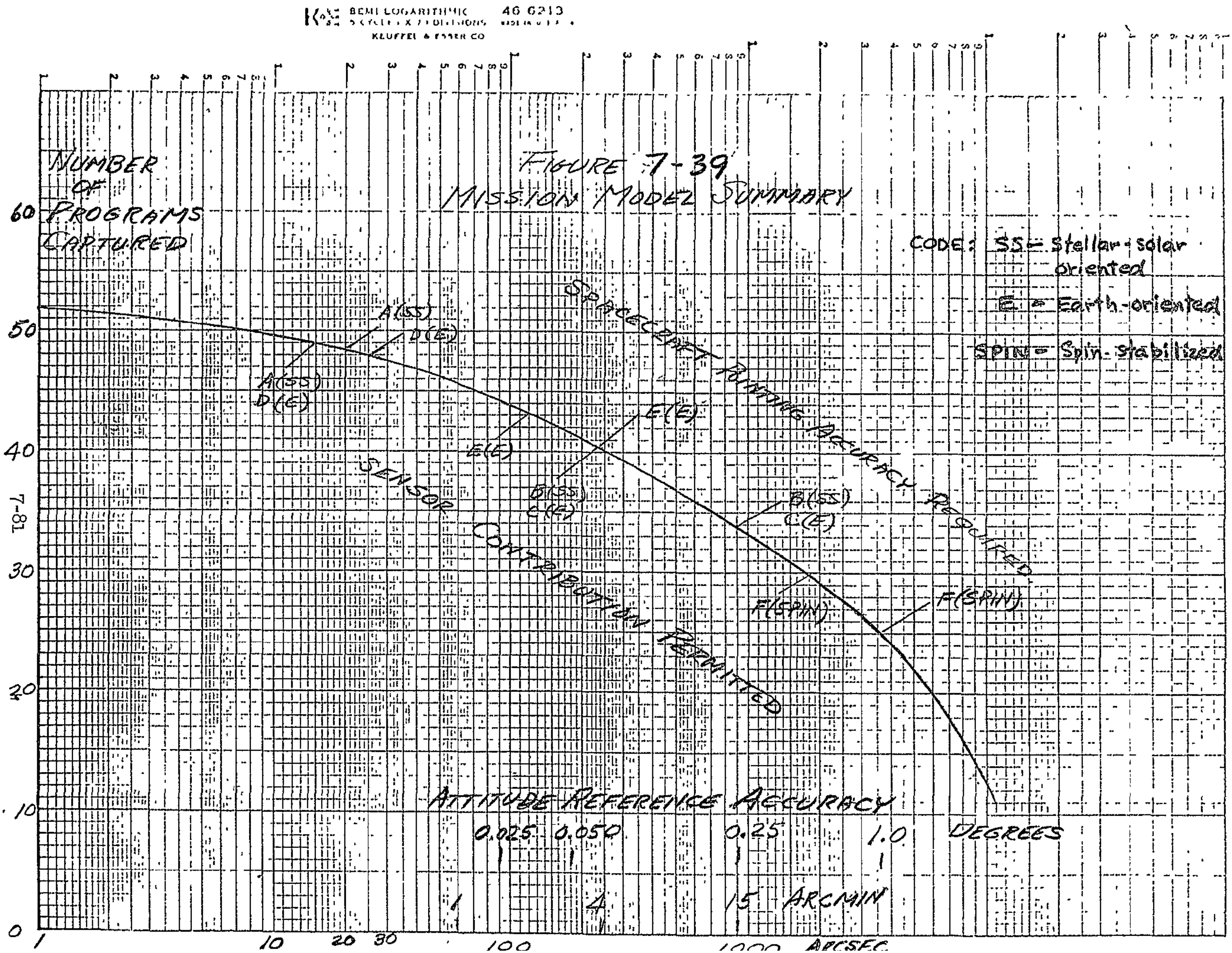
a preselected attitude with a precision as great as 20 $\widehat{\text{sec}}$ (including random payload reference alignment errors of up to 10 $\widehat{\text{sec}}$.

- Orient the spacecraft, as required, to the reference attitudes dictated by the mission objectives.
- Accept attitude pointing error signals from payload sensors with resolution as fine as 0.01 to 0.10 $\widehat{\text{sec}}$ and as coarse as 10 to 15 $\widehat{\text{sec}}$ and control attitude with commensurate accuracy.
- Hold the spacecraft in a reference attitude with the required accuracy over periods of time up to 2 years.
- Reorient the spacecraft to the reference attitude from any attitude following loss of reference due to reversible system failures for tumbling rates up to 10 deg/sec.
- Point the spacecraft at the sun with near zero rates following primary subsystem failure.

Figure 7-39 shows the cumulative number of programs satisfied by a given pointing accuracy. The basic letters (A, B, C -- F) refer to basic subsystem options identified later on Figure 7-41. Figure 7-40 shows the region of applicability of star sensors and horizon sensors.

Within the Mission Model, analysis identified six logical levels of pointing capability, categorized by accuracy, altitude, and spacecraft orientation. Associated with each type is the estimated allowable contribution from the attitude-reference source. If an electronic offset scanning star tracker is used the maximum allowable field of view can be specified. Those conclusions are summarized in Fig. 7-41.

The estimated numbers of multi-use star tracker heads needed vs mission type and attitude pointing requirement are shown in Figure 7-42.


K&E SEMI-LOGARITHMIC 46 6213
5 CYCLES X 70 DIVISIONS
KEUFFEL & ESSER CO.
FIGURE 7-39
MISSION MODEL SUMMARY
NUMBER OF PROGRAMS CAPTURED
CODE: SS = Stellar-solar oriented
E = Earth-oriented
SPIN = Spin-stabilized
SPACECRAFT POINTING ACCURACY REQUIRED
SENSOR CONTRIBUTION PERMITTED
A(SS)
D(E)
A(SS)
D(E)
E(E)
E(E)
B(SS)
C(E)
B(SS)
C(E)
F(SPIN)
F(SPIN)
0
10
20
30
40
50
60
ATTITUDE REFERENCE ACCURACY
0.025 0.050 0.25 1.0 DEGREES
1 4 15 ARCMIN
1 10 20 30 100 1000 ARCSEC

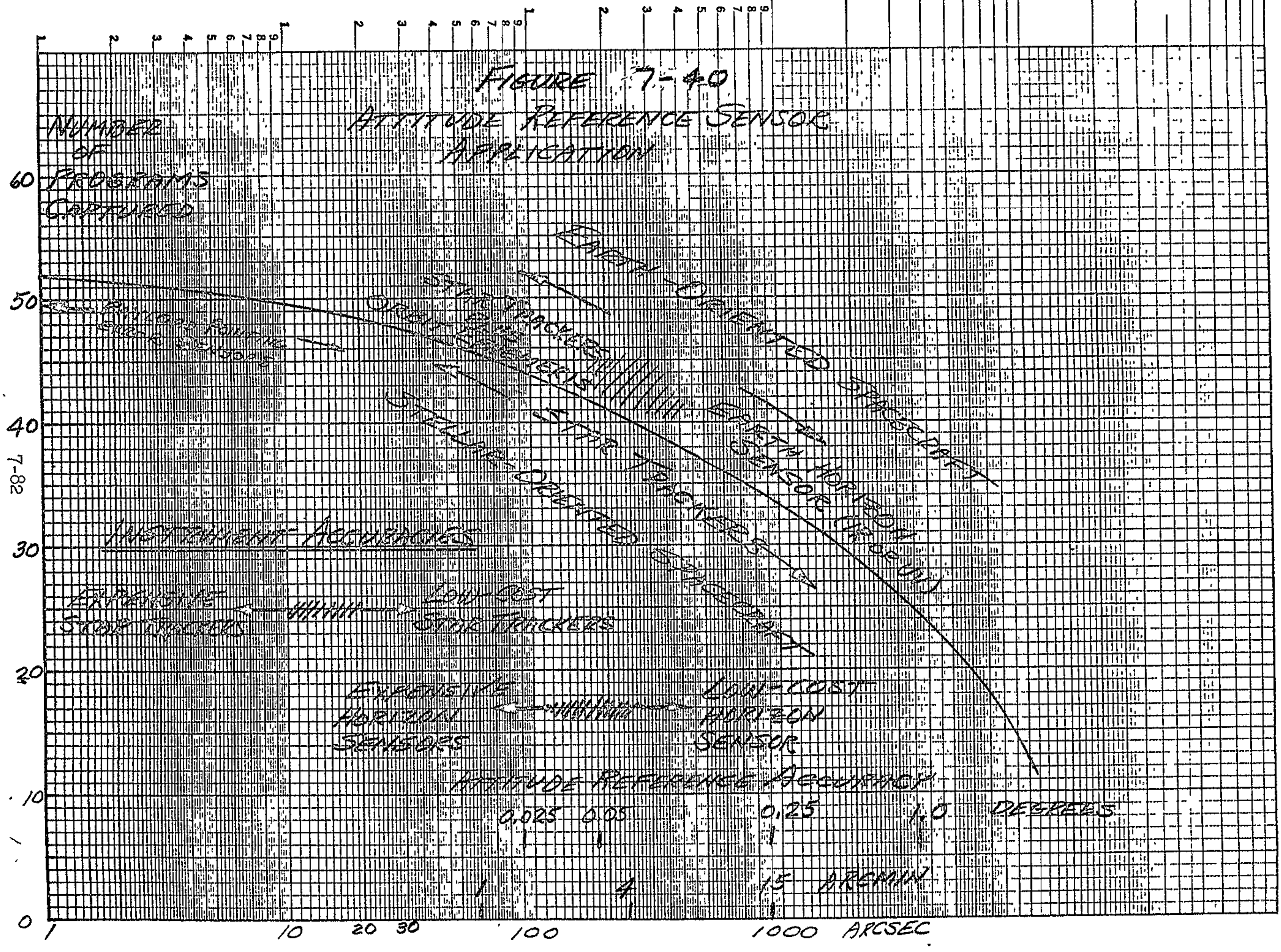


FIGURE 7-40
ATTITUDE REFERENCE SENSOR APPLICATION

System Capability Type	Orientation	Desired Maximum Pointing Error	Estimated Maximum Allowable Instrument Error	Corresponding Field of View (Accuracy: one part in 600)
A	Stellar	20 sec	15 sec	2.5°
B	Stellar	15 min	4 min	40°
C	Earth	15 min	4 min	40°
D	Earth*	25 sec	15 sec	2.5°
E	Earth	4 min	2 min	20°
F	Spin	1 deg	30 min	Not Applicable

* High altitude only.

Fig. 7-41 Pointing Requirement Classifications

		Number of Tracker Heads (Non-Redundant)	
Mission Applications	Required Pointing Precision	Satellite Earth-Oriented	Satellite Stellar-Oriented
Low Earth Orbit	20 sec to 0.25° Less than 0.25°	3 (E) 2*(C)	3 (A) 2 (B)
High Earth Orbit	20 sec to 0.25° Less than 0.25°	3 (D) 1*(C)	2 (A) 1 (B)
Earth Escape	20 sec to 0.25° Less than 0.25°	Not Applicable	2 (A) 1 (B)

* or none - replaced with Horizon Sensor.

Fig. 7-42 Implementation of Requirements with Fixed Head Trackers

7.4.8.2 S&C Subsystem/Implementation. A unit of equipment for multipurpose use should combine a wide range with a high resolution or precision. This ideal is realizable with 1971 technology: The strapdown rate sensor and programmable digital computer have a range of 1 part in 10^5; the digital solar aspect sensor 1 part in 10^4; and the multi-head star tracker 1 part in 10^3. The dynamic range of reaction wheel control torque, depending principally on system quantization, is effectively unlimited.

7.4.8.3 Standard S&C Subsystem Variants and Characteristics. All standard stabilization and control subsystem variants will include the following basic components:

1) Three-axis attitude rate sensor package
2) General purpose digital computer
3) Two two-axis solar aspect sensors
4) Four laser corner cube reflectors
5) Attitude control propulsion drive electronics

The S&C variants will be made up by adding parts selected from the following list of optional equipment:

1) Fixed head star trackers (one to four)
2) Reaction wheels (three)
3) Solar Aspect Sensors (two to six)
4) Wide-range earth horizon sensor (one)

The fixed head star trackers can be fitted with either wide-angle (40 deg max) or narrow angle (≈ 3 deg) field-of-view optics depending upon the accuracy desired. Associated with each star tracker and with the reaction wheels, each pair of solar aspect sensors, the rate sensor package, and the earth sensor are electronics packages. These electronics packages will be mounted in the same modules as their associated sensor or actuator. Their functions will be tailored to produce a low bit rate digital electronic interface with the computer.

The S&C subsystem characteristics are listed on Figure 7-43.

7.4.8.4 S&C Subsystem Operation. The principle of operation of the primary attitude determination function is to obtain accurate long-term attitude information from star sensors and a computer-stored star catalog, while the three-axis rate sensor gyros provide a precise short-term reference. Figure 7-44 shows the functional and equipment relationships. The three-axis rate sensor (TARS) and up to five fixed-head star trackers (FHT) provide attitude data to the computer which combines them in a Kalman filter algorithm to compute spacecraft attitude precisely. For some mission applications, a horizon sensor is substituted for a star tracker.

Attitude control torques are obtained by varying the speed of the reaction wheels or pulsing the attitude control thrusters. The attitude control thrusters are also used to desaturate the reaction wheels by torquing the spacecraft and to slew from one attitude reference to another if required.

The subsystem design approach takes advantage of the repeatability and stability of inertial-grade gyros for measurements of high data-rate attitude changes. Because random variations in gyro parameters are very small (less than 0.01 deg/hr) only discrete updating is necessary. The periodic star or horizon fixes, via the filter in the computer, bound long-term attitude errors and, at the same time, update the estimated random gyro drifts, scale factor errors, and alignment biases.

A check on the vehicle orientation in space is available to the ground station by readout of the digital solar aspect sensors. These same sensors would be used for attitude acquisition and reacquisition should a catastrophic event cause the spacecraft to tumble.

Item	Qty.	Unit Weight (lb)	Size	Power (Watts)	Estimated Failure Rate (per 10 /Hr.)/Duty Cycle %	Potential Supplier
Three-Axis Rate Sensor	1	15	500 in^3	30	8/100	Kearfott
Digital Computer	1	18	150 in^3	25	15/100	CDC
Digital Solar Aspect Sensor	2 (min.) 8 (max.)	0.5	3x3x1 in.	0	0.1/1-100	Adcole
Solar Aspect Sensor Electronics	1 (min.) 5 (max.)	2	5x5x2 in.	2	6/1-100	Adcole
Fixed Head Star Tracker No Electronics	1 (min.) 3 (max.)	12 (typ.)	6Dx12 in. (typ.)	11	5/100	ITT or Ball Bros.
Earth Horizon Sensor & Electronics	1 (Optional)	20	300 in^3	20	6/100	TRW
Reaction Wheel	3 (Opt.)	12	8D x 4 in.	5	1/100	Bendix
Reaction Wheel Drive Electronics	1 (Opt.)	15	10x15x4 in.	9	4/100	LMSC
Attitude Control, Propulsion Drives Electronics	4	7	8x4x3 in.	3	3/25	LMSC
Laser Corner Cube Reflector	4	2	4x4x2 in.	0	N/A	ITT

1 lb = 0.4536 kg

1 $in.^3$ = 16.4 cm^3

Fig. 7-43 Standard S&C Subsystem Parts List

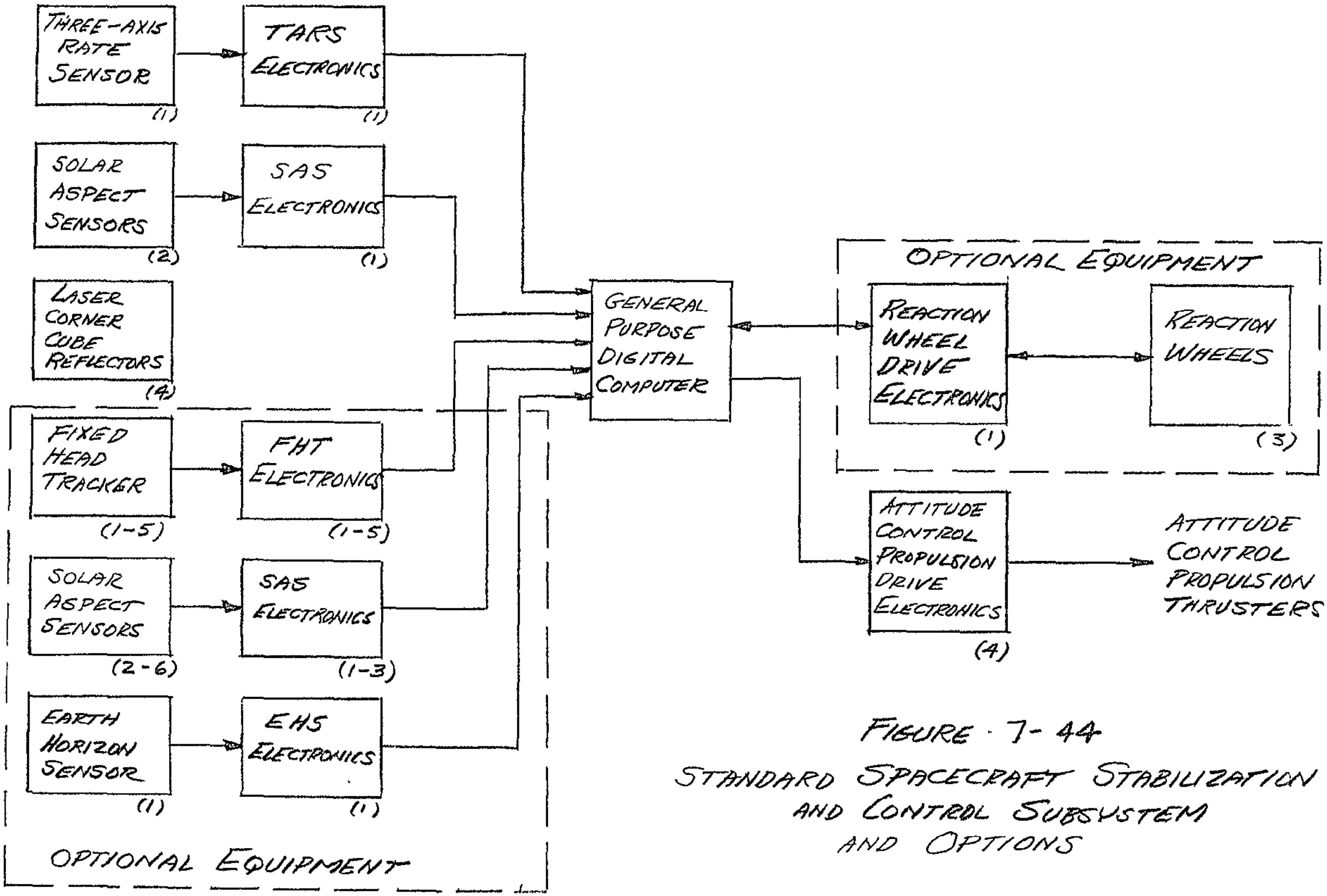


FIGURE 7-44
STANDARD SPACECRAFT STABILIZATION AND CONTROL SUBSYSTEM AND OPTIONS

7.4.8.5 Standard S&C Hardware Elements. The equipment chosen to implement the standard S&C subsystem is flexible. The options available to the user are summarized below.

1) Fixed Head Star Trackers

Field-of-view: Interchangeable optics to maximize number of stars visible for a given accuracy level.

Number of heads: Two to three. Space for five to assure two visible stars, considering earth occultation, sun/moon interference, vacant areas in celestial sphere, and redundancy. Only one head when used as earth UV horizon locator.

2) General Purpose Digital Computer

Memory: Expandable from 4K to 64K 16-bit words

Software: Permits creation of subsystem control laws, sequencing and logic integration to meet program requirements and spacecraft dynamics. Facilitates changes. Sufficient capacity to perform other functions, in particular, data processing and command and telemetry.

3) Reaction Wheels (optional)

Torque: Expandable from 2.0 to 7.5 in-oz. (0.014 to 0.049 newton-meter)

Momentum Storage: Expandable from 0.13 to 6.0 ft-lb-sec (1.76 to 8.15 newton-meter-record) by wheel rim change.

Number: Zero or three depending on disturbance torque environment, mission life, pointing precision, and weight allowance.

4. Digital Sun Aspect Sensors

Field of view: $16^{\circ} \times 16^{\circ}$ to $128^{\circ} \times 128^{\circ}$

Angle Resolution: 1/256 deg to 1 deg

Number: Two to ten depending on coverage desired and accuracy.

5. Earth Horizon Sensor (Optional)

Altitude Range: 100 to 60,000 NM (185 to 111,000 km)

Operating offset capability: $\pm 30^{\circ}$

Accuracy: $\pm$ 0.15 deg

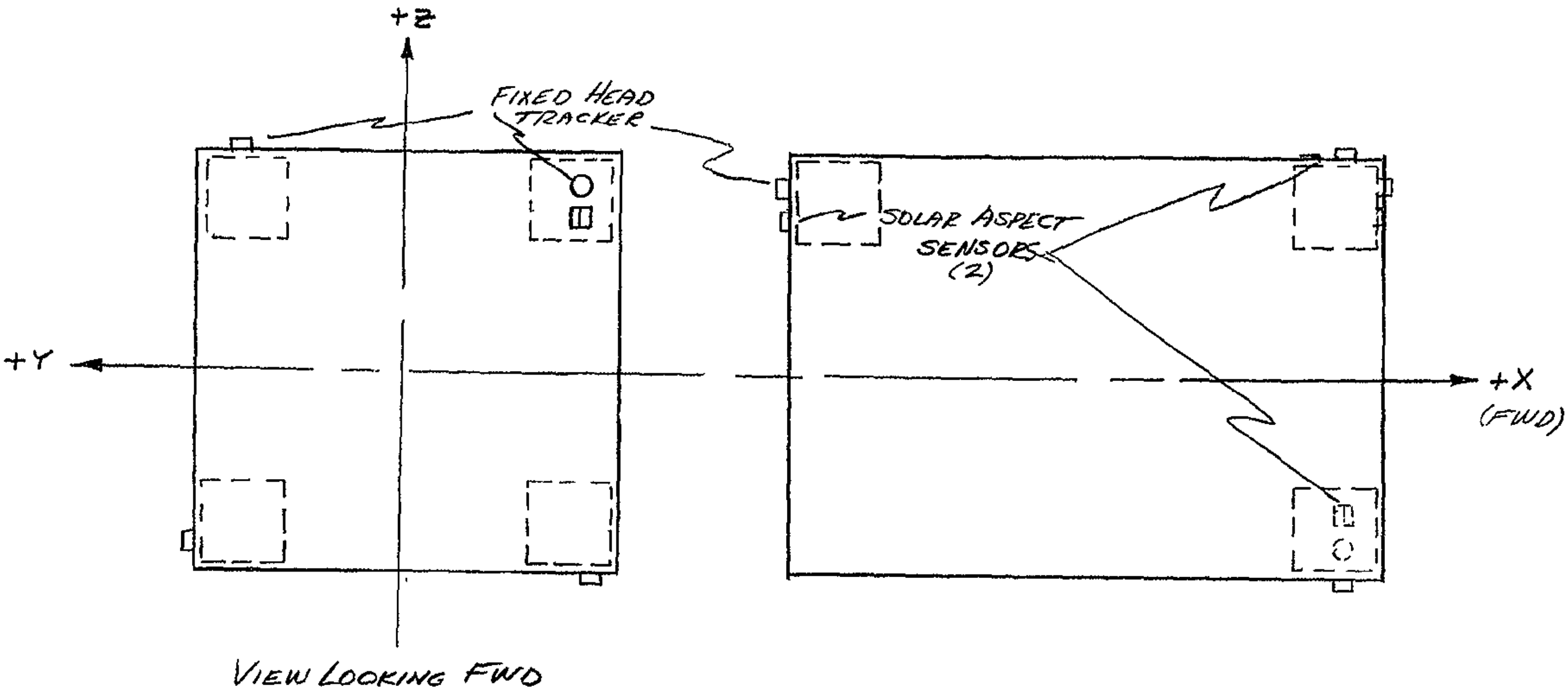
In addition to having the potential of performing as a multi-mode sensor, the fixed head tracker has the capability of being readily adapted to different levels of accuracy. The tracker (photo cathode, image dissector, power supply and sweep control electronics) would be standard, greatly simplifying development testing and the interfaces with the spacecraft. The capabilities, in particular the field of view (FOV) and accuracy, of the tracker would then be established solely by the choice of optics.

The rationale for numbers of trackers other than two rests on: (1) the high incidence of star occultation with low earth orbits; (2) the wider field of view permitted for low precision cases; and (3) the added difficulties with star identification when earth-oriented.

7.4.8.6 S&C Equipment Installation. Figure 7-45 shows one arrangement of S&C sensors. Typically, five bay locations are reserved for identical "Attitude Sensing Modules" although, as indicated earlier, no more than three star trackers are required.

Providing space for up to five fixed head trackers and ten sun sensors permits incorporation of part redundancy for any mission; another Attitude Computation or Reaction Wheel Module could also be added if redundancy is desired in either of these areas.

Each sensing module (Figure 7-46a) has provisions for mounting one fixed head star tracker in either one of two perpendicular orientations, two solar aspect sensors on a variable-incidence mounting bracket and one (optional) horizon sensor and all associated electronics. The remaining equipment is grouped into two other modules (Figures 7-46b and 7-46c); the seven modules are inter-



NOTE: FOR EARTH-ORIENTED APPLICATIONS, +Z IS NADIR

FIGURE 7-45
S & C SENSOR LOCATION OPTIONS

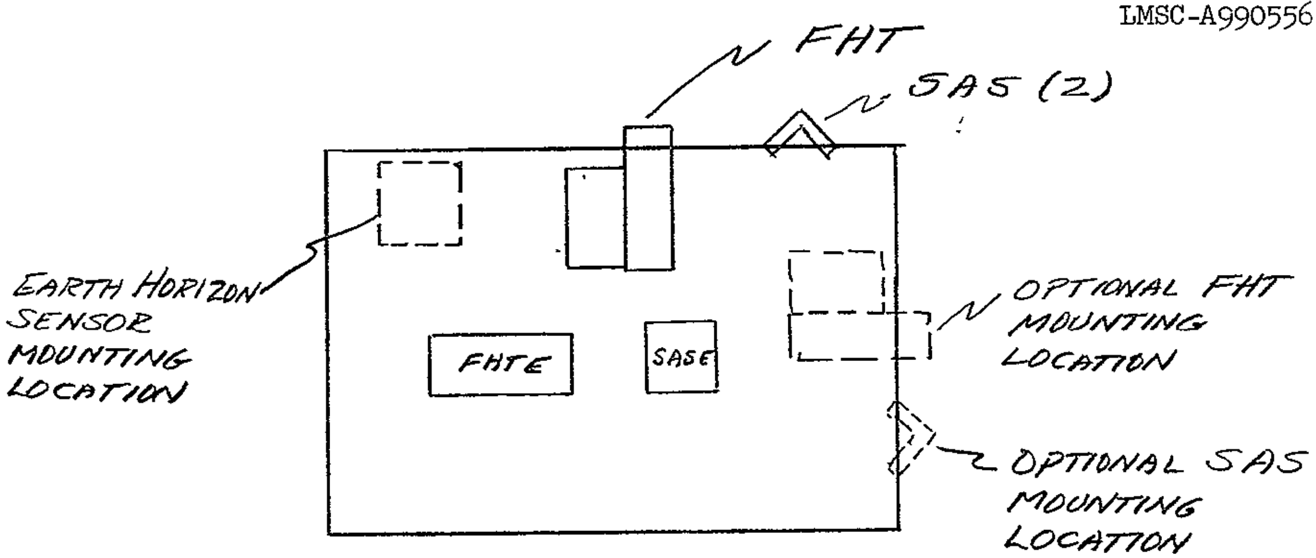


FIGURE 7-46a
ATTITUDE SENSING MODULE (1-5 REQ'D)

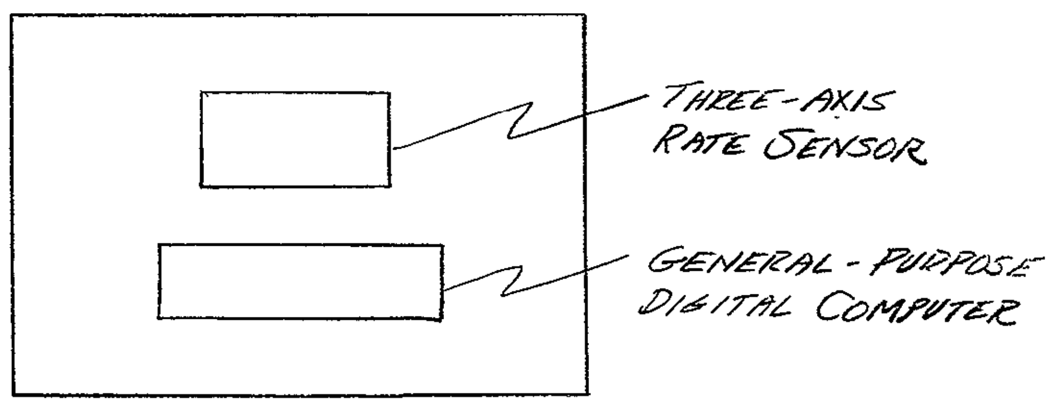


FIGURE 7-46b
ATTITUDE COMPUTATION MODULE (1 REQ'D)

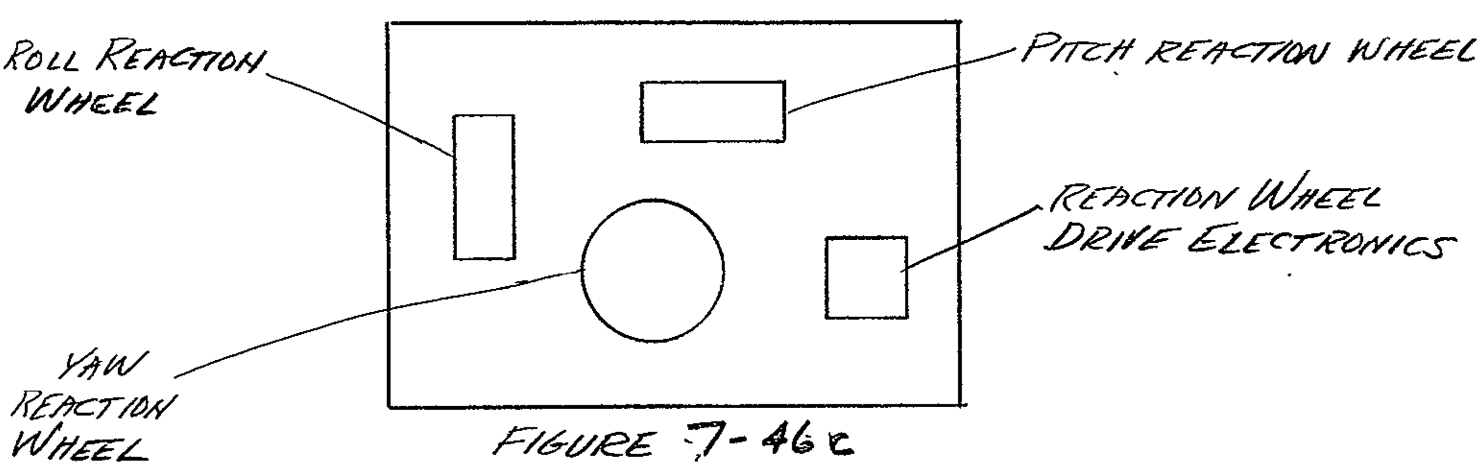


FIGURE 7-46c
REACTION WHEEL MODULE (1 OPTIONAL)

connected as shown in Figure 7-47.

7.4.8.7 Potential Problem Areas in Standardized S&C

a. Development

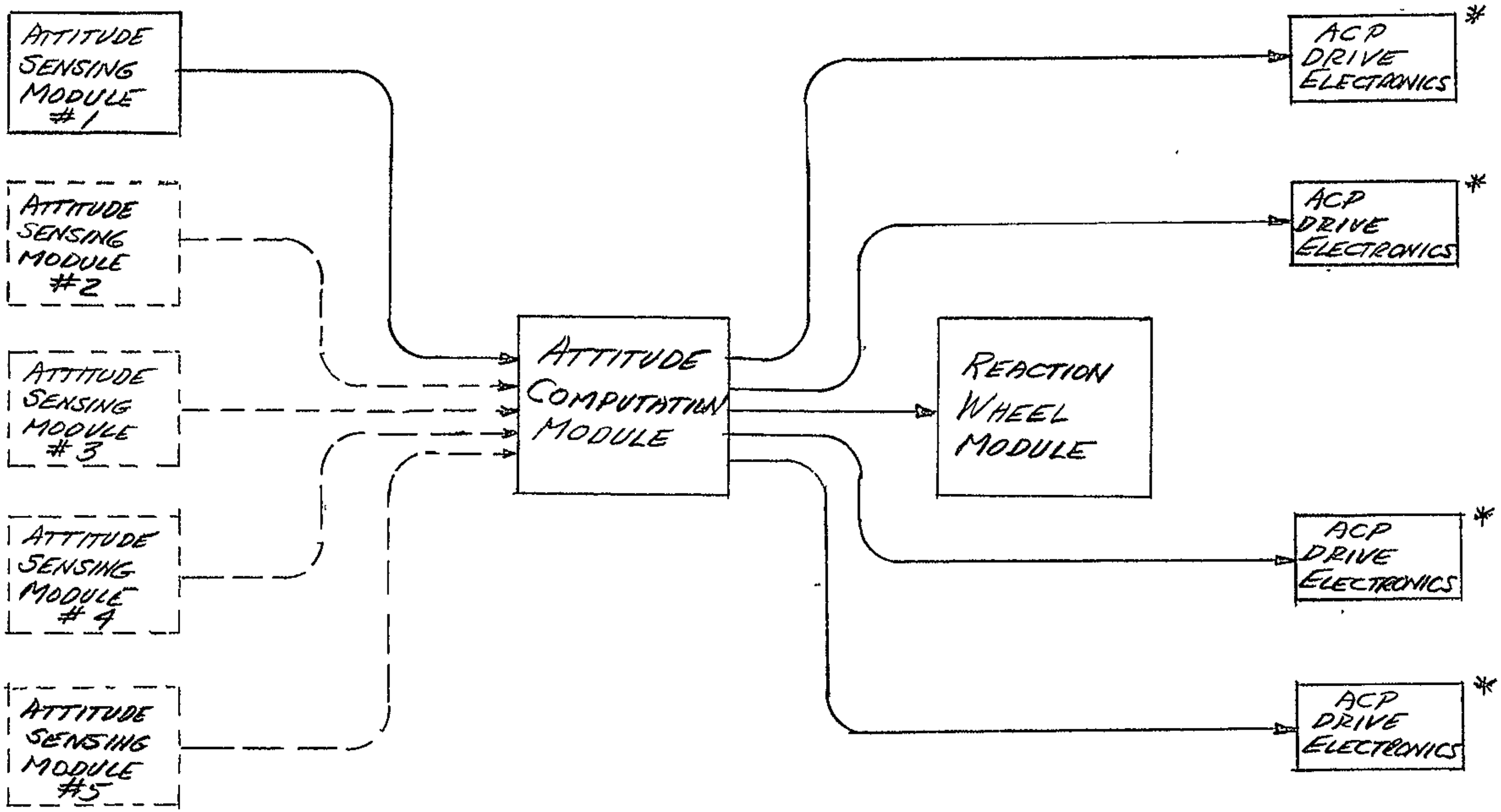
1) To avoid redundant equipment to meet reliability goals, particular attention should be paid to increasing the MTBF of the three-axis rate sensor and the digital computer.

2) Some difficulty has been experienced in obtaining unit-to-unit performance repeatability of electronic star trackers. This problem will persist for high accuracy trackers. Further tracker development will be necessary to consistently achieve the accuracy goal (one part in 600) and, at the same time, realize a unit cost matérially below $100K.

b. Operations

1) Electronic star tracker operation is degraded by stray magnetic fields. It is therefore important to avoid locating sources of EMI near the star sensors and to calibrate them in their actual environment. In the latter situation it may be desirable, and in some cases necessary, to further calibrate the star trackers for the ambient earth magnetic field on orbit and/or provide shielding for the sensors. This implies the possible need for a magnetic field on orbit and/or shielding for the sensors. These considerations imply the possible need for a magnetic simulation facility for spacecraft development and testing.

2) For fine pointing mission spacecraft, the need to assure alignment stability between the star trackers and the spacecraft payload within about 10 arc seconds after spacecraft shipping and handling, and exposure to the launch and orbital thermal environments, will impose important spacecraft design constraints.



* CONTAINED WITHIN ACP MODULE

FIGURE 7-47. S & C MODULE INTERCONNECTIONS

3) To assure initial star acquisition without requiring wide-angle optics, the star tracker-Shuttle GNC reference alignment must be controlled to about 0.25 degrees. This alignment could be held through spacecraft-shuttle adapter design, calibrated by an optical link, or bypassed if the spacecraft computed its attitude from shuttle liftoff.

7.4.8.8 Spin Stabilization. For those programs where a small, spin-stabilized spacecraft is preferred, a different, simpler stabilization and control concept is recommended since only a one-degree pointing accuracy is required. The equipment needed is:

Function	Equipment
Spinup	Cold gas system
Spin axis attitude control	" " "
Spin axis maneuvers	" " "
Nutation damping	Toroidal tube damper
Attitude Sensing	Earth horizon sensor
	Sun aspect sensor
Spin axis maneuvers (optional)	Magnetic torque coil
	Magnetic torquer programmer

An attitude control electronics package completes the implementation list. For programs preferring ground control, the electronics would accept thruster pulse commands from the command decoder. Other attitude control electronics functions are:

a. Process earth horizon sensor outputs to obtain roll angle error and span rate.

b. Process solar aspect sensor outputs to obtain sun elevation and azimuth

c. Supply thruster pulse timing logic

d. Provide backup roll angle and pulse logic in case of one sensor failure.

7.4.9 Summary of Standard Hardware Design Options.

Analysis of the NASA Mission Model concluded that the spectrum of requirements could be satisfied with two standard spacecraft types: (1) a relatively small spin-stabilized and (2) a much larger, three-axis stabilized configuration. Although there are potentially two sets of standard subsystem designs, one optimized for each of these basic spacecraft type; only the three-axis stabilized subsystems have been studied in any depth. The principal characteristics of each standard subsystem; which can be applied separately to any program, or as part of a standard spacecraft; are described following.

7.4.9.1 Electric Power

- 155 to 1110-watt unregulated 28 VDC capability range
- Modular panel, single-axis tracking array 65 ft^2 to 195 ft^2 (6-18 m^2)
- One to six battery-charge controller sets (20 amp-hr)
- Optional regulated 28 VDC and 115 VAC power available

7.4.9.2 Communications and Data Processing

- Up to 10^7 bps PCM from earth orbit and 10^5 bps from 1-3 AU
- 30-foot dia ground antenna for earth orbit link; DSIN for interplanetary
- Digital Telemetry, Tracking and Command using Unified S-Band
- 0.25-w transponder, omni antenna; optional 2.5, 10, and 50 w power amplifiers
- Optional earth coverage, 3-foot, 10-foot, and 30-foot antennas
- Share S&C GP computer to aid data mpx, encoding, formatting, etc.

7.4.9.3 Attitude Control and Stationkeeping Propulsion.

- Option of 8,000 lb-sec or 30,000 lb-sec maximum from four identical modules

- Monopropellant hydrazine, blowdown, passive containment
- Nominal 0.5 lbf and 5.0 lbf thruster option
- Sixteen thrusters and four spherical tanks per spacecraft

7.4.9.4 Stabilization and Control

- Stellar or earth pointing from 20 $\widehat{\text{sec}}$ to 15 mm $\widehat{\text{min}}$
- Strapped down, precision three-axis rate sensor and GP digital computer
- Option of up to 10 solar aspect sensors

7.4.10 Weights of Standard Spacecraft

The gross weight of the three-axis stabilized standard spacecraft excluding experiments but including a 15% contingency will vary from a low of 2600 lb to a high of 4,000 lb* depending upon the performance level. Experiment weights will range from several hundred to several thousand pounds. Smaller experiments than these will be supported by the standard spinning spacecraft, whose design gross weight will vary from about 600 to 900 lb.*

7.5 ECONOMIC EVALUATION OF STANDARDIZED SPACECRAFT/SUBSYSTEMS

7.5.1 Basis for Determining Savings with Standard Spacecraft Hardware.

Standardization of spacecraft elements can be thought of as an extension of low-cost design principles where the emphasis is shifted from the most efficient cost-effective satisfaction of single-mission requirements to the most efficient handling of multi-mission sets of requirements. Low-cost spaceeraft design principles and their associated lower costs, described elsewhere on this report for shuttle-based space operations, are used as the baseline plateau on which the economic evaluation of standard spacecraft approaches shall be made. The cost-savings due to standardization will therefore accrue in addition to those identified previously for implementation of low-cost payloads.

* 1 lb = .4536 kg

7.5.2 Cost Effects of Standardization

Spacecraft standardization, whether implemented at the component, subsystems or spacecraft level, will impact costs in different ways, some of them counteracting others. One of these impacts is the sharing of development funds by a group of missions, the requirements for which can either be met or exceeded by a single standardized hardware development. The program development cost savings increase with the number of sharing applications; however, as a larger set of requirements must be satisfied, it is necessary to provide for an increasing number of interface options. This requires additional systems integration and testing and results in the development costs for a standardized (multi-mission application) item being appreciably higher than for single program-peculiar development.

As to the optimum systems level at which to standardize, the following observations are made. In addition to standardization at the component level, which is most advanced with electronics systems, it is intuitively evident that standardization at a higher systems level would avoid repetition of a larger portion of the systems integration costs; i.e., the potential savings would increase with increasing systems-level integration if it were not for the testing expense of super-integration. A conclusion in this regard cannot be drawn from the consideration of development costs alone. The desired approach must be associated with minimum total program costs and therefore must include unit cost of standardized equipment as another variable. If a given spectrum of requirements are to be covered by a small finite number of equipment options, it is inevitable that requirements are "overkilled" by a factor which is inversely related to the available number of discrete options the practical simplification of this, however, is not as severe as it might appear. In most subsystems, with the exception of electrical power (EPS), the cost figures cluster around a plateau, i.e., the unit cost increment for performance overkill is likely to be small. There are two more aspects which seem to mitigate the costs effects of overkill; these are: (1) the discounting of future recurring expenditures in terms of excess unit cost and (2) the consideration of

spacecraft residual value and reuse (in which case excess unit cost is pro-rated over a number of missions).

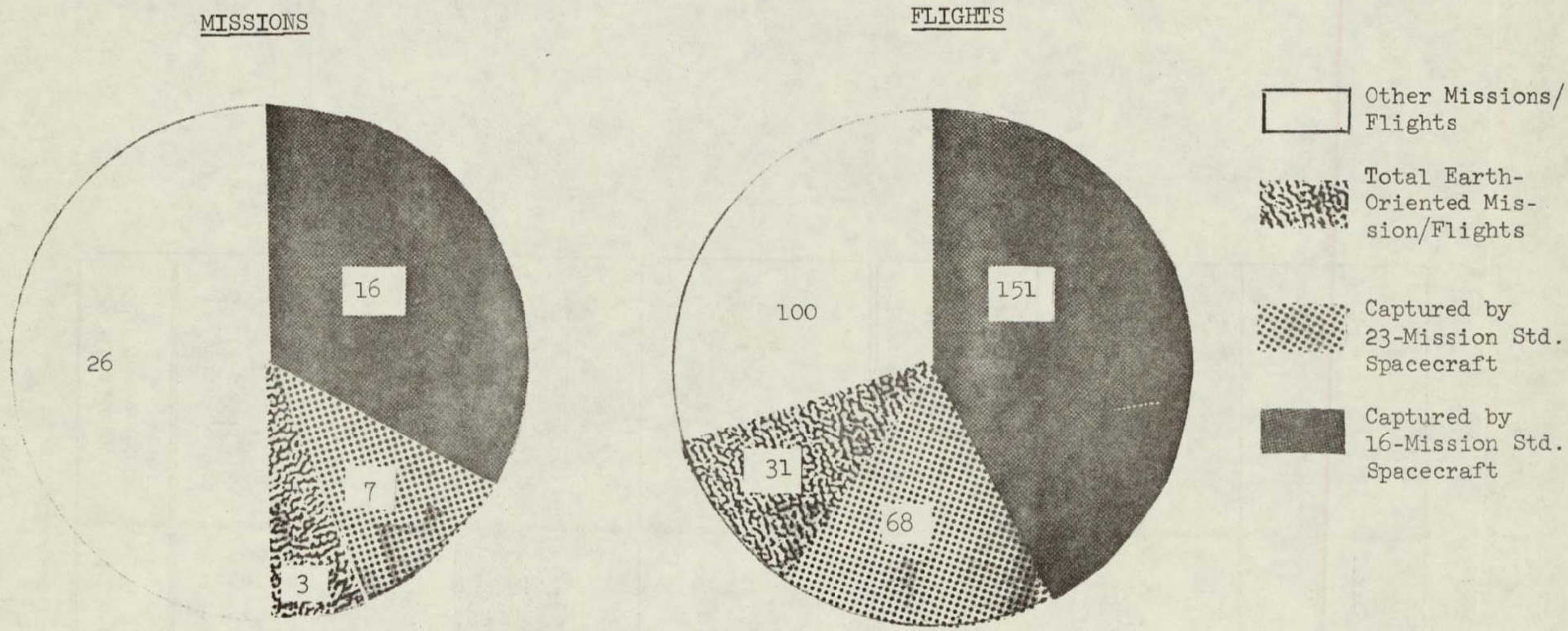
7.5.3 Economic Analysis

In the standard spacecraft study task, the emphasis was not on optimization, but on the identification of gross cost savings that may be realized from spacecraft standardization. Two specific examples are offered to illustrate the effectiveness of different approaches:

(1) The first case investigated involves a multipurpose spacecraft which is configured to satisfy the requirements of a large segment of the mission model. The cost savings over the mission-peculiar development approach are evaluated for that segment of the mission model.

(2) The second case investigated involves the application of standardized and modularized subsystems to the majority of the the unmanned payload programs in the mission model.

7.5.3.1 Savings with Multipurpose Standardized Spacecraft. A survey of the NASA Mission Model showed that a large number of the earth-oriented low and synchronous orbital satellite missions could be accommodated by a single multipurpose spacecraft design with somewhat overdesigned subsystem capabilities. This approach would require only a fraction of the hardware developments that would be required to cover individual mission-peculiar developments. As shown in Figure 7-48, a spacecraft designed to capture 16 missions will be duplicated in 151 flights. It contains CDPI for high data rate transmissions from synchronous orbit, has a horizon sensor GNSC, and contains EPS and APS to accommodate the maximum requirements. The cost evaluation in Figure 7-49 shows that standard spacecraft RDT&E expenditures drop to less than 10% of the program-peculiar approach. However, there is a penalty paid in terms of unit costs due to the aforementioned requirements overkill (resulting in higher unit cost for standardized hardware than for project-peculiar).



23-Mission Spacecraft - Star-Trackers for precise pointing reference plus maximum capability in other subsystems

16-Mission Spacecraft - Horizon Sensor for pointing reference plus maximum capability in other subsystems

Fig. 7-48 Missions/Flights Captured by Earth-Oriented Standard Spacecraft

Mission Coverage	Sub-System	Sub-System Option	RDTE ($ Million) Program Peculiar	RDTE ($ Million) Standard Spacecraft	Unit ($ Million) Program Peculiar	Unit ($ Million) Standard Spacecraft	Standard Spacecraft Savings ($ Million)
16 Missions 151 Spacecraft	CDPI	C	211.5	18.0	307.4	422.8	
	GNSC	C	76	9.5	280.9	280.8	
	EPS	C	212.8	17.4	282.7	350.3	
	ACS	B	29	2	39.1	39.3	
	Struct.	-	61	5.3	65.1	90.6	
	ECS	-	22.0	1.8	25.1	30.2	
	Totals		612.3	54.0	1000.3	1214.0	344.6
23 Missions 219 Spacecraft	CDPI	C	327.50	18.0	479.0	613.2	
	GNSC	E	157	27	598.5	1022.7	
	EPS	C	309.8	17.4	416.1	508.1	
	ACS	B	39	2	56.4	56.9	
	Struct.	-	102.4	7.3	113.1	175.2	
	ECS	-	35.8	3.0	43.0	65.7	
	Totals		971.5	74.7	1706.1	2441.9	161.0

Fig. 7-49 Economic Impact of Application of Typical Standard Spacecraft

Disregarding a potential amelioration due to discounting of future costs and possible spacecraft reuse, the evaluation shows that $344.6 millions can be saved by use of this standardized spacecraft. In view of this encouraging result it was of interest to see whether additional savings could be made by increasing the capability of the multipurpose spacecraft. It was found that by incorporation of startrackers into the GNSC system, 7 additional missions could be captured which, according to Figure 7-48, involved an additional 68 flights to bring the total up to 219 flights. As is evident from Figure 7-49, the penalty is terms of additional unit costs increases out of proportion to the savings from RDT&E sharing, so that the total program savings are reduced to only $161 millions, or less than half of the 16-mission standardization. Generalized to the total traffic model, this approach may be promising if the requirements spectrum is subdivided into compatible mission sub-sets for which the marginal savings due to RDT&E sharing exceed the cumulative unit-cost expense of requirements overkill as indicated in Figure 7-50. Additional considerations of spacecraft refurbishment/reuse or dollar-discounting may be added to aid in deriving optimized cost-effective combinations of missions.

7.5.3.2 Savings with Modularized Standard Subsystems. Another basic approach to satisfying Mission Model requirements is to build up program-peculiar spacecraft from an inventory of standardized subsystems options.

The standard subsystem options used for economic evaluation were defined in sub-section 7.4. ACS and EPS were assumed to be standardized at the subsystems level. CDPI and S&C were standardized below the subsystems level in a way that performance-capability options are achieved by the addition/deletion of components within a basic standard subsystem.

Figure 7-51 shows the subsystem equipment options used to cover almost the entire mission model. Also listed are the associated development costs using the program-peculiar and the standardized approach. It shows that more than a 20:1 reduction in effective development cost may be achieved using standard

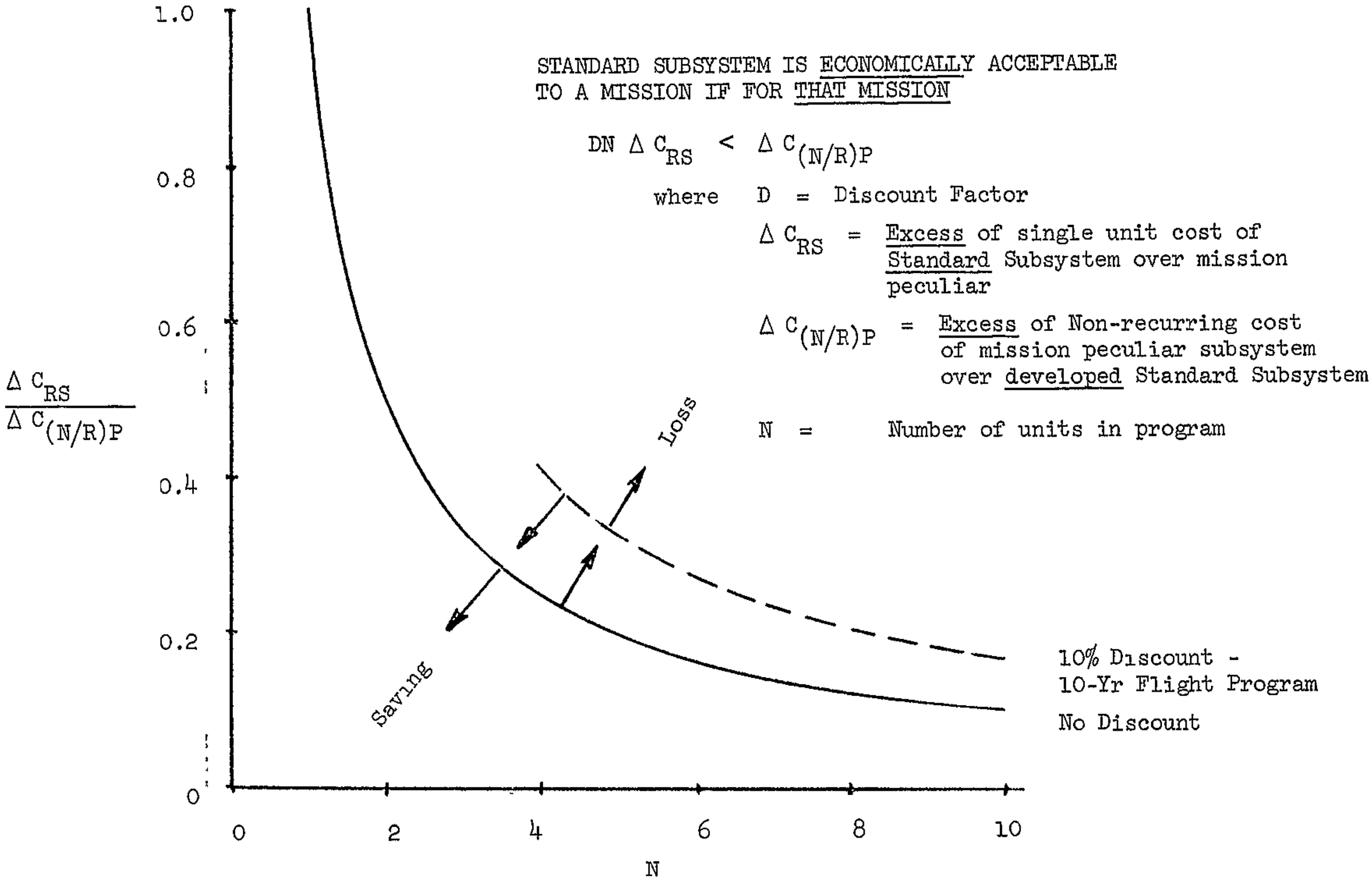


Fig. 7-50 Criterion for Acceptability of Standard Subsystem to Specific Mission

Subsystem	Option	Qty. Programs	RDTE ($ Million)		
			Program Peculiar		Standard Total
			Per Program	Cumulative Total	
CDPI	A	13	$ 38.0	$ 494	-
	B	4	18.0	72	-
	C	12	18.0	216	-
	D	3	5.5	17	-
	E	12	15.5	186	-
	COMSAT	8	6.5	52	-
	Total	52	-	$1037	$58
GNSC	A	9	27.0	243	-
	B	13	9.5	124	-
	C	8	9.5	171	-
	D	5	1.3	7	-
	E	8	27.0	216	-
	Total	53	-	$ 761	$32
EPS	A	16	10.6	169	-
	B	14	14.4	202	-
	C	6	17.4	104	-
	D	4	6.7	27	-
	Total	40	-	$ 502	$24
ACS	A	17	1.0	17	-
	B	31	2.0	62	-
	Total	48	-	$ 79	$ 3
			GROSS TOTALS →	$2379	$117

Fig. 7-51 Economic Impact of Application of Standard Subsystems

subsystems. It should be noted that in this case the degree of excess unit costs is minimal because the amounts of "overkill" for each standard subsystem application is small (the several options of standard subsystems allows choice of one very close to the program-peculiar capability requirement).

However, there is a recurring requirement for spacecraft integration which must be accounted for. In view of the fact that discounting and spacecraft reuse must be considered also in this context it is not possible to note at this time whether the standard subsystems approach is cost-wise superior to the multipurpose standard spacecraft approach. Nevertheless, it is important to note that by either approach considerable amounts of money (and development time) can be saved, and that further more detailed study of the subject cannot fail to identify an even larger savings potential.

Section 8
LOW-COST PAYLOAD INTERFACES WITH SPACE TRANSPORTATION SYSTEM

The influence of the expendable launch vehicle and Space Shuttle upon the low-cost payloads has been evaluated and the interfaces between the payload and the transportation vehicles have been studied and defined. Primary emphasis was placed upon the payload/Shuttle interfaces because the expendable launch vehicle interfaces essentially do not differ from those which have existed historically for a number of years.

The general effects of the launch/ascent environment upon the payloads have been analyzed. The highlights are discussed in sub-section 8.1.

A new concept of checkout of payloads on-board the Shuttle has been developed and is described in sub-section 8.2. The implementation of this concept is considered vital to the cost-reduction approach involving pre-deployment check-out and on-orbit repair/refurbishment.

Because of the strong influence of deployment and retrieval hardware upon payload configuration and structural design, a special universal-usage deployment gear concept was developed to the extent that low-cost payload interfaces could be validated. The configuration and functions are described in sub-section 8.3.

The most significant cost-driver element among the low-cost payloads is the repair, refurbishment, and reuse of payload hardware. The total concept and potential cost reductions are described in sub-section 8.4.

8.1 EFFECT OF SHUTTLE ENVIRONMENT ON PAYLOADS

During the development of the low-cost payload preliminary designs, consideration was given to effects of the Shuttle environment. The Phase B Contractor

data, supplied via Aerospace Corporation to LMSC have been revised during the study period, most recently as of 22 March 1971. In general, there was no profound effect upon payload design. A qualitative assessment of these effects was made; the general results are summarized in the following paragraphs.

8.1.1 Load, Shock/Vibration, and Acoustic Effects

The latest data, which place the Shuttle launch/ascent and reentry loads at 3g maximum, provide potentially a softer ride of the payload on the Shuttle than on the new low-cost expendable launch vehicles. However, payloads mounted flexibly onto the Shuttle structure or suspended cantilever-style from a support platform (like the historical payloads mounted atop a booster vehicle) will probably be exposed to load-amplifications which will approach those with the expendable launch vehicles. It is important therefore that structural mounting of the payload within the Shuttle cargo bay be given primary attention to obtain minimum net loads upon the payload.

The vibration conditions resulting from Shuttle engine transients have not yet been fully specified. The load level and frequency will determine the feasibility and efficacy of attenuation devices in mounting of the payloads.

The acoustic environment of the Shuttle, calculated at the external skin surface has been estimated at 158.5 db OASPL. However, the attenuation (or amplification) by the Shuttle structure of this energy (relevant to the payload in the cargo bay) has not been determined.

Although the Shuttle data available are somewhat qualitative in nature, the design approach used on the low-cost payloads makes the payloads essentially insensitive to reasonable levels of shock, vibration, and acoustic excitation. The payload structures, designed with high safety factors (3 or greater), utilize comparatively thick external skins, large cross-section beams, and ruggedized equipment module mountings. Equipment, mounted on rigid platforms within the modules, is all but isolated from the mechanical environmental effects.

8.1.2 Thermal Environment Effects

The temperatures predicted for the Shuttle cargo bay are in general within the limits of those experienced historically by payloads mounted within exit fairings on expendable launch vehicles. The maximum temperature of 150°F (339°K), occurring during launch/ascent and reentry, can be tolerated quite well by the low-cost payloads. It is possible, with the Shuttle cryogenic propellant tanks mounted adjacent to the cargo bay cavity, that ground ventilation and/or cooling during the pre-launch cycle will not be necessary. Also, because Shuttle equipment mounted in the vicinity of the cargo bay also will be as sensitive as the payload to elevated temperatures; any general ventilation or forced air flow provided can also be used to minimize temperature rises in the payload. This most critical phase probably will occur after landing where the residual heat in the Shuttle thermal protection materials may gradually raise the internal Shuttle temperatures until the cargo bay doors can be opened.

In general, the Shuttle temperature environment does not appear to offer a specific problem for the unmanned payloads studied.

8.1.3 Pressure Variation Effects

The pressure in the Shuttle cargo bay during launch/ascent will decrease from 14.7 psi (1.013×10^5 newton/m^2) to essentially space vacuum in about 120 seconds. Conversely, the pressure will increase in the reentry/landing mode from space vacuum to one atmosphere in approximately 1600 seconds. Although this pressure change will not affect the majority of payload hardware, special care has been taken in design of the low-cost payloads in the following areas:

a. Tanks

For tankage which usually starts the mission charged with pressurized gas or fluids and wherein depletion during orbit operations is possible, provision must be made to sustain the external pressure during reentry by (1) over-designing the tank shell to carry the negative pressure, (2) retaining

residual pressure in the tank, or (3) re-pressurizing the tank for the return flight to earth.

b. Closed Volumes

For equipment which is packaged in essentially a closed box or compartment, both in and out air flow must be provided unless the element is designed sufficiently strong to sustain separately both an internal and external pressure of one atmosphere. It should also be noted that the reentry/descent operation will force air into the previously air-void compartment; contaminants and water vapor will also enter unless the volume is sealed or otherwise protected. This condition is little different from that which has been encountered on high-altitude aircraft in past years. If composite structures such as honeycomb-stiffened panels are used, similar precautions are necessary in providing perforated materials to allow air leakage out (on ascent) and water vapor release (after descent).

In general, no problem was encountered with the low-cost payload design in this study: (1) No honeycomb structure was utilized; (2) Basic structure was designed to sustain the effects of internal or external pressure applications; (3) Components within the equipment modules have essentially sealed housings to prevent moisture and contaminant entry and all exposed surfaces are designed for exposure to water vapor conditions; (4) Module structural covers are designed to "bleed" air fast enough through screened orifices to prevent positive or negative pressure buildup.

8.2 SHUTTLE ON-BOARD CHECKOUT OF PAYLOADS

The concept of standardized interfaces between payloads and the Space Shuttle cargo bay may be extended to include a payload test set carried on board the Shuttle and used for monitoring and checkout of one or more payloads. Such a test set may be independent or it may utilize some of the Shuttle computational, data handling and display services. In either case it must interface with the Shuttle data bus and supply safety status information on the payload(s) to the

Shuttle flight deck. The configuration and the capability of the Shuttle integrated avionics system to support payloads has not yet been established, nor have the detailed requirements for computer support, control and display for checkout of the spectrum of shuttle payloads been fully identified. Payload cost savings have been estimated, based on this study, to accrue from the use of on-board payload checkout and are sufficiently significant to warrant further investigation.

8.2.1 Hardware Elements Requiring Checkout

Typical unmanned payloads consist of a spacecraft and one or more experiments, and may be categorized in terms of weight, size, spacecraft functions, orbital parameters and desired life. Included are both expendable and reusable propulsion stages or tugs, and combinations of spacecraft and tugs. For the purpose of on-board testing, each major payload or element of a group of payloads may be synthesized into 8 basic functional groupings or subsystems as portrayed in Fig. 8-1. Individual elements within the group change in complexity and detail from program to program, or even flight to flight, as shuttle cargo mixes vary; this requires tailoring of the checkout procedures and the modules that make up a standard test set for each situation or mission. The similarity of function and limited range of parameters does permit a high degree of standardization and/or modularization of the checkout equipment as well as the spacecraft components.

8.2.2 Basic On-Board Checkout Concept

A basic premise for on-board checkout (OBC) is that the payload(s) has been completely tested and calibrated by subsystem and function, and has been accepted as flight-qualified or flight-ready before it is delivered to the shuttle cargo loading facility. The primary purpose, therefore, of OBC, is to verify that the payload(s) still functions after experiencing rather severe environmental changes and that its critical safety, survival and operational parameters are within pre-determined limits. Monitoring or sampling the instrumentation subsystems provide status data; applying stimuli, including commands,

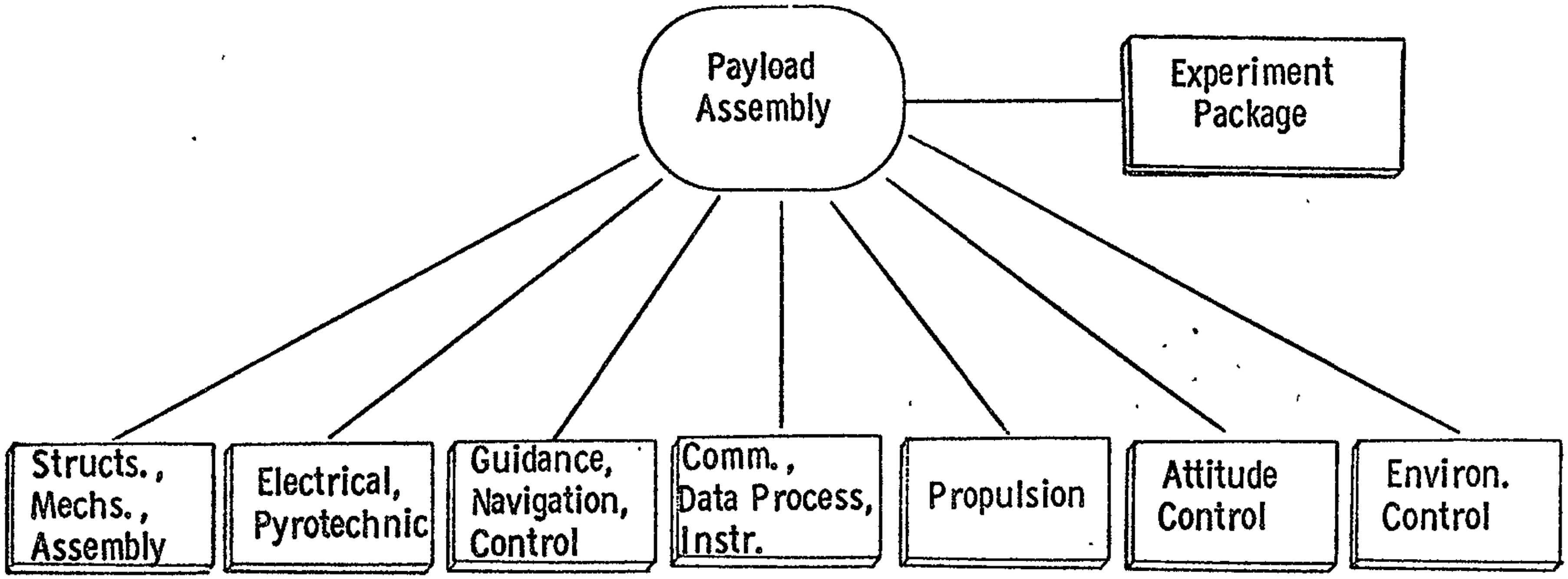


Fig. 8-1 Typical Payload Subsystem Groupings

and comparing end-to-end system response to pre-stored data confirms configuration and operation. A Payload Test Set, necessary for OBC, may function, therefore, as the electrical and data interface between payloads and the orbiter cargo bay junction boxes and standard digital interface units that provide access to the orbiter data bus.

8.2.3 The Payload Test/Checkout Set

A payload checkout or test set (PTS) capable of monitoring and detecting hazardous conditions, status of critical items and configuration, and verification of system functions typically contains the elements shown in Fig. 8-2. A preliminary estimate of the physical characteristics of a PTS is shown in Fig. 8-3; Fig. 8-4 compares this proposed Shuttle-carried checkout set with a typical ground checkout station made up of typical ground-based commercial hardware elements (the Shuttle-carried set employs solid-state electronics of the 1970 state-of-the-art). Allowance was made for an ancillary checkout computer roughly comparable to a non-redundant version of the UNIVAC 1832 used in the S3A aircraft, modified to reduce power required. Used for testing low-cost payloads designed to interface with it, the PTS can have a capability comparable with existing large, dedicated ground checkout complexes.

The checkout function differs from the monitoring function in that the equipment or subsystem under test must be in a known condition or configuration, with known inputs or stimuli; the function is usually characterized by a command/response sequence. Hence, the PTS elements must be completely compatible with the system being checked out. For example, if the payload employs a 149 MHz command frequency the PTS command transmitter must generate a properly modulated 149 MHz carrier; another payload on the same flight might use an S-band system, thus requiring an additional command module in the PTS. Similarly, the PTS telemetry modules may be called upon to handle VHF or S-band links employing various modulation techniques and data rates and to provide decommutated outputs in compatible digital format that can be handled by the PTS computer. The PTS, therefore, should provide for multiple plug-in command, telemetry, stimuli, and power units to avoid having to carry more than one PTS on some

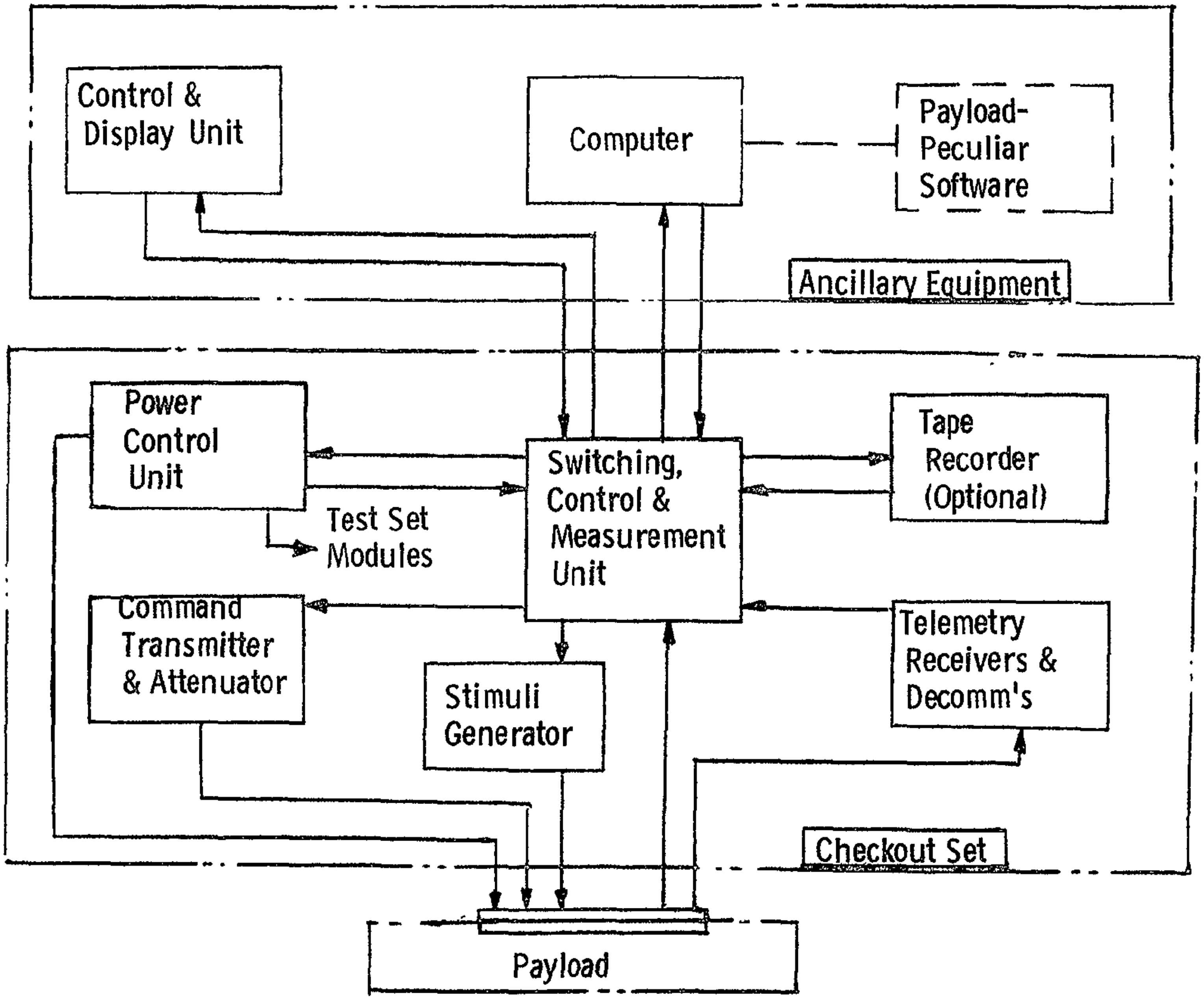


Fig. 8-2 Block Diagram - Payload Test Set

Hardware Element	Size (in.)	Weight	Power Req'd.
(1) Switching, Control, Meas. Unit	14 x 19 x 23	40 lb.	60 W
(2) Power Control	7 x 19 x 23	30	30
(3) Command Transmitter/Attenuator	6 x 19 x 8	20	30
(4) Stimuli Generators	9 x 19 x 23	40	35
(5) T/M Receivers/Decomms	6 x 19 x 23	20	25
(6) Test Umbilical	-	20	-
(7) Tape Recorders	11 x 19 x 23	40	40
(8) Control/Display (External)	(13 x 19 x 23)	40	30
(9) Computer (External)	(ea. 11 x 19 x 23)	200	300
TOTALS	≈ 72 x 19 x 23	450 lb	550 W

1 in. = 2.54 cm
1 lb = 0.4536 kg

Fig. 8-3 Preliminary Size/Weight Estimate - Payload Test Set (for Use in Shuttle)

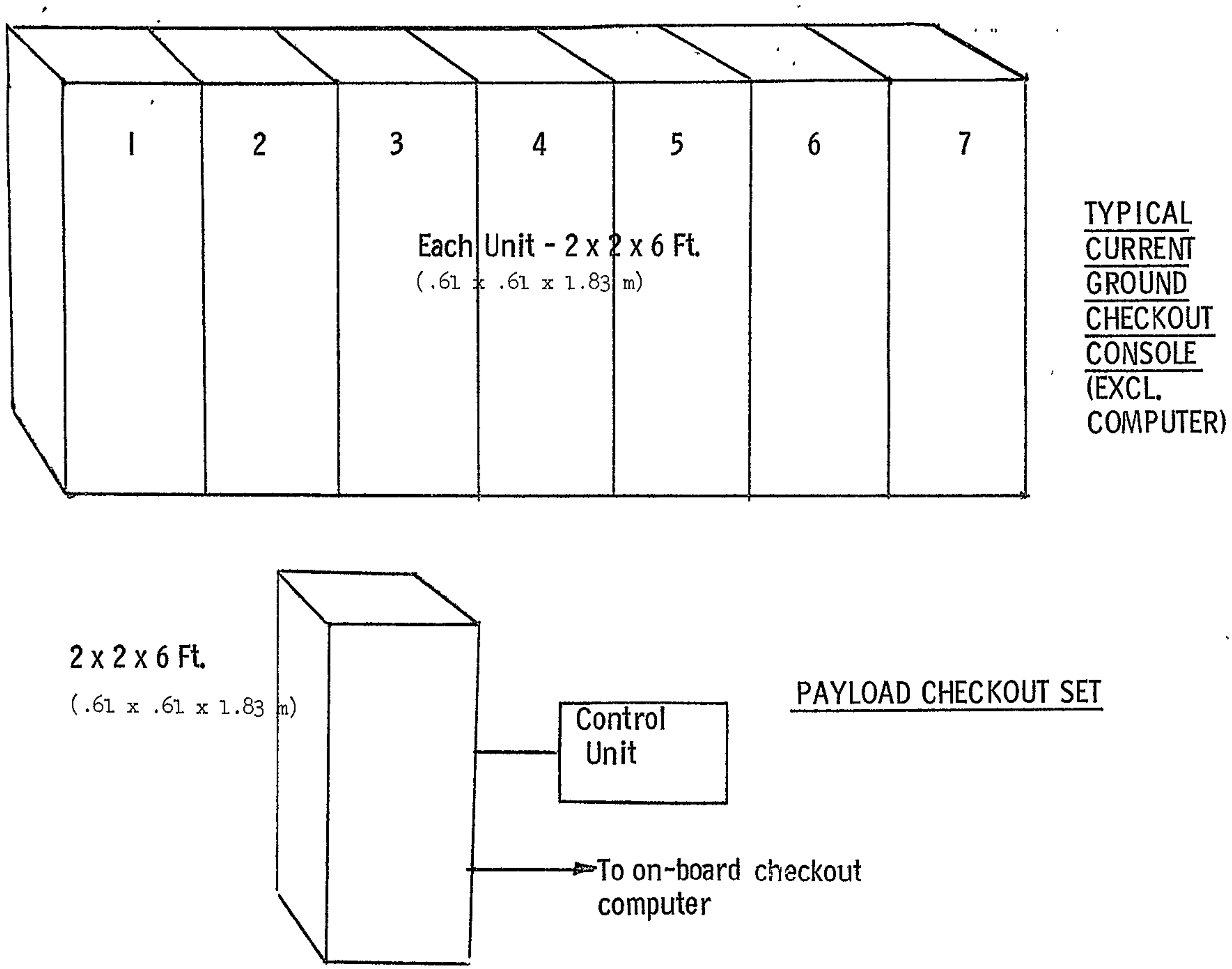


Fig. 8-4 Payload Checkout Set Comparison

shuttle flights. Software for checkout, including the computer programming, is peculiar to each payload, but with many common sub-routines. A powerful, test-oriented programming language and a fully developed basic program is considered a necessity, so that generation and modification of the specific payload-peculiar control program is relatively simple. An oversized memory is essential so that programming may be accomplished economically and to permit real time program modifications in response to on-line checkout situations. Test procedures and initial values and limits for specific tests are established during the payload development and qualification phases and confirmed or modified during experiment integration and acceptance test phases. The functional tests performed on-board are simplified end-to-end tests, usually under nominal conditions, and are "operational" rather than "R&D" or "engineering" in nature. On the first flight of a given payload, and with a mission flight plan that permits a delay of several revolutions in LEO before deployment, engineering tests may be performed to augment the test data base; while these tests undoubtedly have appreciable value, they should be considered a bonus rather than prime mission design drivers.

8.2.4 On-Board Checkout Operations

OBC could be completely automatic, controlled by stored programs, without direct participation by the shuttle crew; if a large number of completely repetitive missions were to be flown and the extra "payload crew" were not needed for other activities such an approach might prove desirable. However, most missions involve unique payloads, or the interval between flights is long and the payloads are not static, so the human computer may contribute significantly to the checkout process, in addition to being available for IVA or EVA corrective action, including payload module replacement (of a failed module determined by checkout). Considerable work remains to be done in defining man's role in this type of activity.

For purposes of shuttle performance comparison, typical flight profiles call for ascent into a 45 or 50 x 100 nm (83 or 93 x 185 km) transfer orbit,

circularization into a 100 nm (185 km) parking orbit, remaining for several revolutions, injection into a higher transfer orbit, placement or rendezvous maneuvers, deployment or retrieval, return to 100 nm (185 km) circular, and reentry and landing. Typical times in the transfer orbits are 40 to 55 min and in the parking orbit 88 min per revolution. Such coast or orbiting modes are periods of low demand upon both shuttle systems and crew, and are ideal for conducting OBC. If the payload is singular and very simple, checkout may consist of a scan of the instrumentation readouts and a sample command/command-verification test, requiring perhaps five to ten minutes. More complex payloads, such as represented by the low-cost OAO-B, involving a number of modes and redundant systems, may require a half-hour to an hour for pre-deployment readiness verification. Multiple payloads, such as the SEO-Tug combination, may need anywhere from a half-hour to two hours, with an hour suggested as a nominal on-orbit checkout time. On-orbit maintenance activity based on OBC, involving manual repair or module replacement by means of IVA or EVA, obviously requires longer spans, necessitating a parking orbit mode.

8.2.5 Checkout Phases

The ability to perform payload checkout on-board the Shuttle provides the opportunity and advantages of a consistent series of up to five launch and flight operations test modes or phases. Using the same PTS, separate tests on the payload (the integrated systems test and pre-mate readiness tests) can also be performed. This use of a constant set of test parameters provides malfunction and failure trend data particularly important for payload programs involving a relatively few flights stretched over a long time span.

Following a cursory analysis of typical unmanned payload test and checkout functions, a phased methodology was developed. Figures 8-5a through -5g list the basic test/checkout functions to be performed in each of seven phases.

1. COMPLETE FUNCTIONAL TEST, AMBIENT CONDITIONS

   ELECTRICAL POWER
   INSTRUMENTATION
   COMMUNICATIONS
   DATA PROCESSING
   STABILIZATION & CONTROL
   GUIDANCE & NAVIGATION
   ATTITUDE CONTROL
   EXPERIMENT PACKAGES

2. REPEAT FUNCTIONAL TEST AFTER EXPOSURE TO ACOUSTIC LAUNCH/ASCENT ENVIRONMENT SIMULATION

3. REPEAT FUNCTIONAL TEST IN ALTITUDE CHAMBER - HI TEMPERATURE

4. REPEAT FUNCTIONAL TEST IN ALTITUDE CHAMBER - LO TEMPERATURE

5. REPEAT FUNCTIONAL TEST, AMBIENT CONDITIONS

Note: The above "Acceptance Tests" provide the data base for diagnosis, trend analysis and the establishment of nominal values and limits.

Fig. 8-5a Phase 1 - 1st Integrated System Test of Payload Package (at Payload Assembly Site)

1. PAYLOAD/TEST SET INTERFACES
2. TEST SET SELF-CHECK
3. INSTRUMENTATION READOUT AND CALIBRATION
4. ELECTRICAL POWER SYSTEM
5. COMMUNICATIONS SUBSYSTEM
6. DATA PROCESSING SUBSYSTEM
7. GUIDANCE, NAVIGATION, STABILIZATION & CONTROL
8. ATTITUDE CONTROL
9. EXPERIMENTS
10. WEIGHT & CENTER OF GRAVITY DETERMINATION
11. MECHANICAL "SHUTTLE INTERFACE" CHECKS

Fig. 8-5b Phase II - PMRT - Pre-Mate Readiness Tests (at Launch Site)

1. PAYLOAD/SHUTTLE INTERFACE CHECKS
2. PAYLOAD TEST SET SELF-CHECK
3. INSTRUMENTATION STATUS
4. ELECTRICAL POWER SYSTEM (INCLUDING SHUTTLE POWER)
5. COMMUNICATIONS SUBSYSTEM (EMI COMPATIBILITY)
6. DATA PROCESSING SUBSYSTEM (WITH SHUTTLE DATA BUS AND DISPLAY SYSTEM)
7. ATTITUDE CONTROL SYSTEM FLIGHT READINESS CHECKS
8. ALIGNMENT OF PAYLOAD WITH RESPECT TO SHUTTLE

Fig. 8-5c Phase III - PCST - Payload Combined Systems Test (Payload Installed in Shuttle)

1. PAYLOAD/SHUTTLE INTERFACE CHECKS
2. PAYLOAD TEST SET SELF-CHECK
3. INSTRUMENTATION STATUS
4. ELECTRICAL POWER SYSTEM CHECK
5. COMMUNICATIONS SUBSYSTEM CHECK
6. DATA PROCESSING SUBSYSTEM CHECK (WITH SHUTTLE)
7. STABILIZATION & CONTROL ALIGNMENT AND CHECK
8. ATTITUDE CONTROL SYSTEM STATUS CHECK
9. EXPERIMENT CHECK
10. INITIALIZATION OF PAYLOAD FOR ASCENT

Fig. 8-5d Phase IV - PLRT - Pre-Launch Readiness Test (Payload in Shuttle on Launch Pad)

1. PAYLOAD/SHUTTLE INTERFACE CHECKS
2. PAYLOAD TEST SET SELF-CHECK
3. INSTRUMENTATION STATUS
4. ELECTRICAL POWER SYSTEM TESTS
5. COMMUNICATIONS TESTS (ALL MODES)
6. DATA PROCESSING SYSTEM TESTS
7. STABILIZATION & CONTROL OPERATIONAL TESTS
8. ATTITUDE CONTROL SYSTEM STATUS CHECK
9. EXPERIMENT TESTS
10. INITIALIZE PAYLOAD FOR DEPLOYMENT

Fig. 8-5e Phase V - PDRT - Pre-Deployment Readiness Test (on Orbit in Cargo Bay)

1. ELECTRICAL POWER SYSTEM TESTS (SOLAR ARRAYS)
2. COMMUNICATIONS TESTS (WITH TRACKING STATIONS)
3. STABILIZATION & CONTROL TESTS
4. ATTITUDE CONTROL TESTS
5. INSTRUMENTATION STATUS
6. EXPERIMENT OPERATIONAL TESTS
7. INITIALIZE PAYLOAD FOR SEPARATION

Fig. 8-5f Phase VI - PSRT - Pre-Separation Readiness Test (Payload Deployed/Not Released)

1. COMMUNICATION TESTS WITH SHUTTLE AND STADAN
2. ELECTRICAL POWER SYSTEM TESTS
3. ATTITUDE CONTROL SYSTEM TESTS
4. STABILIZATION AND CONTROL SYSTEM TESTS
5. EXPERIMENT OPERATION TESTS
6. INSTRUMENTATION STATUS
7. INITIALIZE PAYLOAD FOR FIRST EXPERIMENT OPERATION

Fig. 8-5g Phase VII - OOT - Operational On-Orbit Tests (Payload Separated - Shuttle Near-By)

8.2.6 Validation of On-Board Checkout Concept

Test documents from the OAO, OGO, and OSO programs provided by NASA/GSFC were analyzed and tested for compatibility with the LMSC-developed checkout concept of a standard payload test set and Shuttle on-board checkout. The results, discussed following, indicate that the concepts described above are feasible and practical.

8.2.6.1 OAO Checkout. In addition to discussions with the OAO Project Office and Test & Integration personnel at NASA/GSFC, computer printouts of the following test procedures for OAO-3 were obtained (from the NASA/GSFC historical data files):

- S-C 3 Countdown to Launch Procedure, Tape 461
- Stabilization & Control Subsystem Functional Test, Tape S&C-60
- Functional Test for ATV, Tape GEP-60
- Communication Functional-60, Tape 141
- Data Processing Functional
- Power Subsystem-60 Complete Functional Test

These tapes were studied and used to evaluate and modify the details of the seven test phases portrayed in Figs. 8-5a through 8-5g and confirm the requirement for the PTS described in par. 8.2.3. An expansion into detail test listings was made for Phases III through VII. Figure 8-6 shows a sample of this expansion for tests of Phase V, Pre-Deployment Readiness Test, for the low-cost OAO payload in the Shuttle cargo bay, on orbit. The difficulty of providing external stimuli for some payload functions while within the shuttle, and the necessity for testing these functions were considered; no valid reason could be found for carrying anything but electrical stimuli for Shuttle on-board testing, particularly with natural stimulation available in Phases VI and VII.

In summary, it was determined that the OAO-3 could be adequately checked out using a Shuttle-carried payload checkout set.

TEST	EXECUTION TIME	REMARKS
V.3 INSTRUMENTATION STATUS	2 MIN.	PART OF THE CDP&I SUBSYSTEM PURPOSE IS TO VERIFY THAT CONDITIONS ARE WITHIN PREPLANNED TOLERANCES. INCLUDES 12 PRESSURE, 90 TEMPERATURE, 24 CURRENT, AND 14 VOLTAGE MEASUREMENTS. ACTUAL READOUT REQUIRES ONLY ABOUT 7 SECONDS. ONLY OUT-OF-TOLERANCE CONDITIONS ARE DISPLAYED.
V.4 ELECTRICAL POWER SYSTEM TESTS	4 MIN.	CHECKS ALL POWER SYSTEMS EXCEPT SOLAR ARRAYS FOR ASCENT SURVIVAL.
4.1 CHECK EACH Ni-CD BATTERY (3)		53 INSTRUMENTATION POINTS.
4.2 CHECK STATE-OF-CHARGE UNIT & DIODE BOX		PTS DUMMY LOADS.
4.3 CHECK POWER CONTROL UNIT		
4.4 CHECK POWER REGULATOR UNIT		
4.5 CHECK BOTH VOLTAGE REGULATORS AND CONVERTERS		
4.6 CHECK BOTH INVERTERS, INCLUDING AUTOMATIC SWITCHOVER		
4.7 COMMAND #1 BAT ON AND VERIFY POWER TRANSFER		
4.8 CHECK POWER DISTRIBUTION UNIT		
4.9 CHECK LIGHT/DARK SWITCHOVER		
4.10 VERIFY ALL BATTERIES FULLY CHARGED		

Fig. 8-6 Low-Cost OAO Phase V PDRT (Sample Excerpt)

8.2.6.2 OGO-F and OSO-E1 Checkout. To further determine the applicability of these Shuttle on-board checkout concepts to other payloads, the following documents, made available by NASA/GSFC, were thoroughly reviewed:

OGO-F	Final Article Test Plan, TRW No. 02648-6040-T0000
OSO-E1	Launch Stand Payload Checks, BBRC TN 65-325
OSO-E1	Thermal Vacuum Test Package, BBRC TN 65-319
OSO-E1	Comprehensive Acceptance Test Procedures, BBRC TN 65-318

The test programs outlined in these references were, of course, developed for the expendable launch mode, with no second chance available, such as provided by on-orbit module replacement or retrieval. Extensive use was made in these ground test programs of reptitive testing and special external stimuli such as sun guns, spin flashers, magnetic devices, radiation sources and externally applied test aids to supplement the umbilical and RF links. Such measures were considered necessary to obtain the very high confidence levels of system functioning and optimization required for launches on expendable boosters. Payloads designed for shuttle launch do not require as good a reliability or as high a confidence level because they can be repaired or returned if not working satisfactorily after ascent and on-orbit checkout. For these reasons, the ground test programs may be reduced by employing simpler tests and fewer iterations, with consequent RDT&E cost savings.

For example, the historical OGO-F countdown included checks of: (a) spacecraft RF commands, data handling, power systems and experiment operation; (b) payload configuration for the terminal count; (c) the terminal count, and (d) an abort procedure. A data center, an RF van, and peripheral equipment were employed.

A payload checkout set in the shuttle cargo bay is capable of conducting the essential parts of the OGO-O-110 Abbreviated Functional Test, including - with the assistance of shuttle maneuvers - response to ACS stimulation. Experiments would be in a quiescent state under these conditions. Further, with the payload deployed on booms from the shuttle, the experiments could be activated

and checked out by the Shuttle onboard checkout set before handover to ground control; handover capabilities can be verified prior to payload separation. In other words, the payload checkout set, the shuttle payload crew, and the Shuttle ground communications link replaces the RF checkout van, the ground test crew, and the ground communications and data links. This approach allows the OGO-F control center to optionally exercise the same command/control analysis functions as before.

The OSO-E1 is characterized by a multiplicity of experiments requiring many types of external stimuli to completely check out. The spacecraft, separately, however, can be rather thoroughly controlled and tested through RF or umbilical links connected to the Shuttle-carried checkout set. In particular, the T-0 Day Checks may be made with the payload checkout set onboard the shuttle, as well as the T-1 Day Checks of Power System, Communications System, Command response, Configuration and Launch Status.

As with OGO-F, the potential limitations in checkout of OSO-E1 are the use of external stimuli other than those available through Shuttle maneuvers or from natural sources in orbit with the Shuttle cargo bay doors open. Since the payload is presumed to have been tested and totally operational before it is loaded into Shuttle, and is carefully handled and monitored from then on, the repeated application of external stimuli does not appear necessary nor desirable. Deployed operation of experiments and sensors may be verified after the payload has been extended on booms from the cargo bay and still attached to the Shuttle. Checkout in this mode may be through an umbilical, by RF link to the Shuttle or directly under control (RF) of ground stations.

Some modifications to the OGO and OSO would, of course, be required for adaptation to Shuttle and to use of Shuttle onboard payload checkout. These primarily involve provisions for: (1) the test umbilical, (2) RF switches for closed-loop testing and control, (3) remotely activated safe-arm switches instead of manual safe-arm plugs, and (4) payload safing for aborts or recovery modes. Similar provisions will also be required for all other payloads to be used with the Shuttle.

In summary, the OGO-F and OSO-E1 payloads could be checked out onboard the Shuttle with a payload checkout set (similar to that concepted for the OAO-B) before deployment and separation in orbit.

8.2.7 Operational and Cost Effects of On-Orbit Checkout by Shuttle

The primary advantages of use of on-orbit checkout of payloads are summarized following:

a. Greatly Increases Probability of Mission Success

Checkout of the payload after the launch/ascent ride and replacement of degraded-performance or failed equipment eliminates a relatively large percentage of launch/ascent failures. Provides a "launch base on orbit".

b. Allows First-Hand Observation of Payload Function on Orbit

The payload-cognizant personnel, whether on the Shuttle or on the ground, can read payload data and add special verification tests as required to verify status of payload before committing to orbit deployment.

c. Makes Feasible On-Orbit Module Replacement and Re-Checkout

Rather than return the payload to earth, with attendant exposure to re-entry, landing, and another launch/ascent, it is desirable to maintain the payload in its relatively benign orbit environment. Replacement of payload modules on orbit (repair or refurbishment) accrues direct benefit. Also, two or more payloads can be "serviced" in orbit by a single Shuttle launch which carries a multiple quantity of replacement modules; this provides direct reduction in transportation costs (compared to round trip transport of the single payloads to earth and return to orbit).

8.3 DEPLOYMENT/RETRIEVAL OF PAYLOADS BY SHUTTLE

The method of support of a payload in the cargo bay of the Shuttle, and the devices used for checkout, deployment, and retrieval of the payload, have a distinct effect upon the payload structural configuration and the arrangement of extendable booms, solar arrays, and other equipment. It was necessary, therefore, to investigate the structural/mechanical interfaces between the three low-cost payloads and the Shuttle to assure that the proposed payload designs were feasible and practicable. Layouts were made of the payload installations in the Shuttle and a universal deployment/retrieval gear was conceptually designed to the point where compatibility with the low-cost payloads was demonstrated.

This sub-section provides descriptions of the OAO, SEO, and SRS installations in the Shuttle and summarizes the characteristics of a universal deployment/retrieval gear. This is not stated to be the only approach which would be suitable for use with various unmanned payloads; rather, it is a single concept used to verify typical Shuttle interfaces with the Shuttle.

8.3.1 Installation and Deployment/Retrieval of the Low-Cost OAO

The installation of the low-cost OAO in the Shuttle is illustrated in Fig. 8-7. A cradle assembly, equipped with a drogue funnel and latch at each of its four corners, accepts four support hold-down pins (detail shown on Fig. 8-8). The cradle is in turn latched down to 2L/2R supports rigidly attached to the Shuttle cargo bay structure; these four mounting points are the mechanical interface between the OAO and the Shuttle and all launch/ascent, maneuvering, reentry, and landing loads are transferred through these points.

Upon attainment of orbit position and following completion of OAO checkout within the cargo bay, the cradle hold-down pin latches are electrically actuated to "open", releasing the payload/cradle from the Shuttle. Four extendable booms (one at each corner of the cradle) raise the OAO from the cargo bay cavity (shown in Fig. 8-9). The solar array paddles are unfolded.

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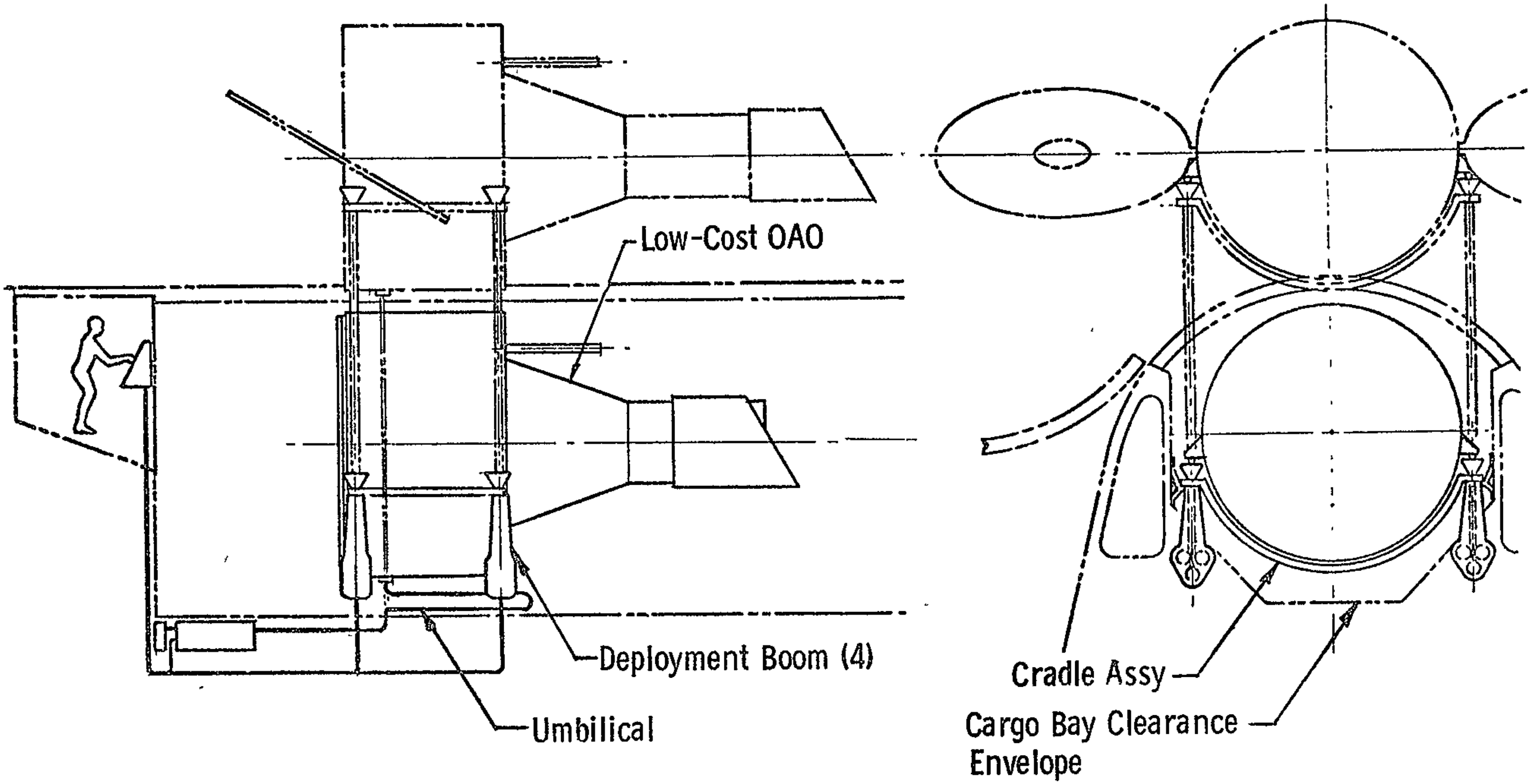
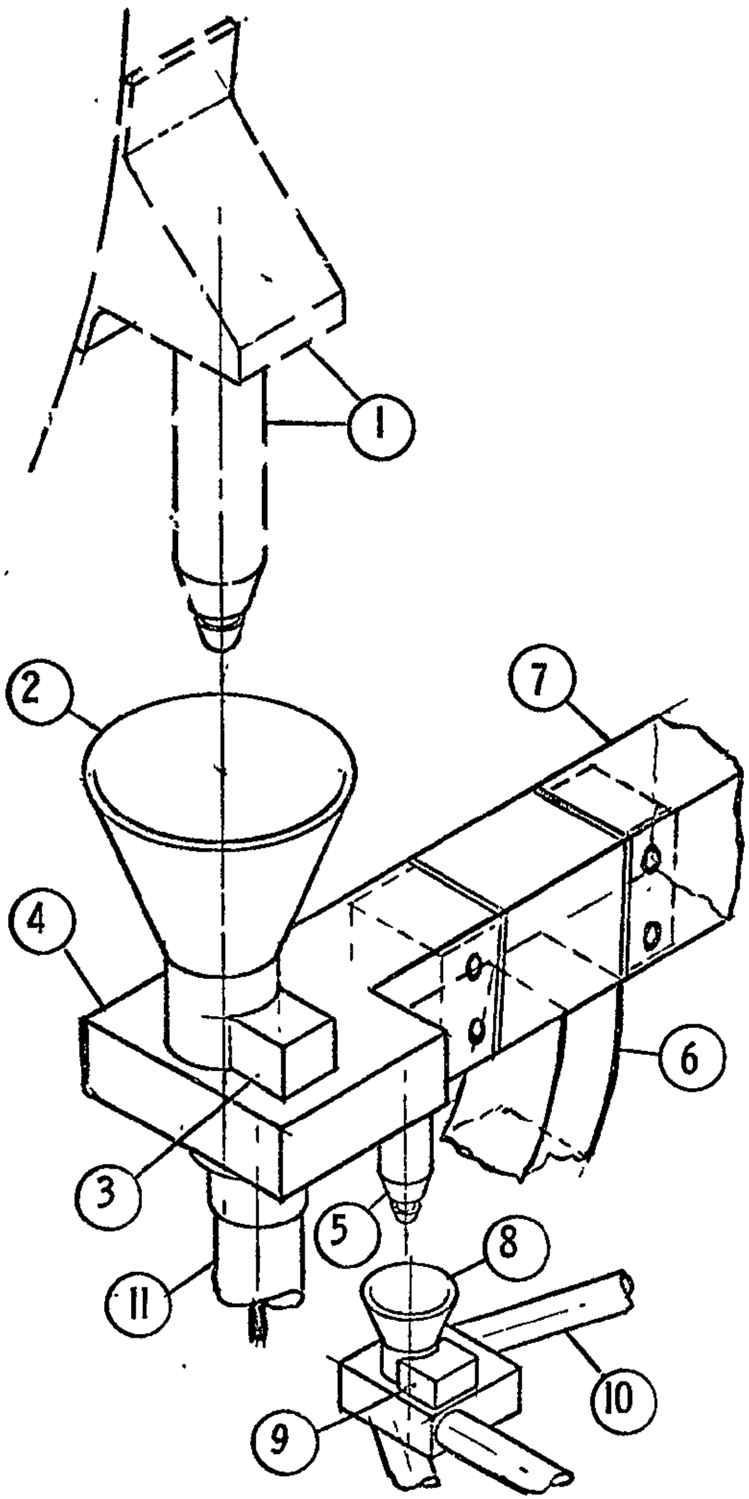


Fig. 8-7 Shuttle Installation of Low-Cost OAO



1. Support Lug and Hold-Down Pin-Payload (Ref.)(4)
2. Drogue Funnel (4) - Payload/Cradle
3. Latch - Cradle Assy (4)
4. End Fitting - Cradle Assy. (4)
5. Hold-Down Pin - Cradle (4)
6. Cradle (2)
7. Spreader Bar (2)
8. Drogue Funnel - Cradle/Shuttle (4)
9. Latch - Shuttle (4)
10. Support - Cradle/Shuttle (4)
11. Bi-Stem Boom (4)

UNIVERSAL PAYLOAD DEPLOYMENT/RETRIEVAL MECHANISM ARRANGEMENT

Fig. 8-8

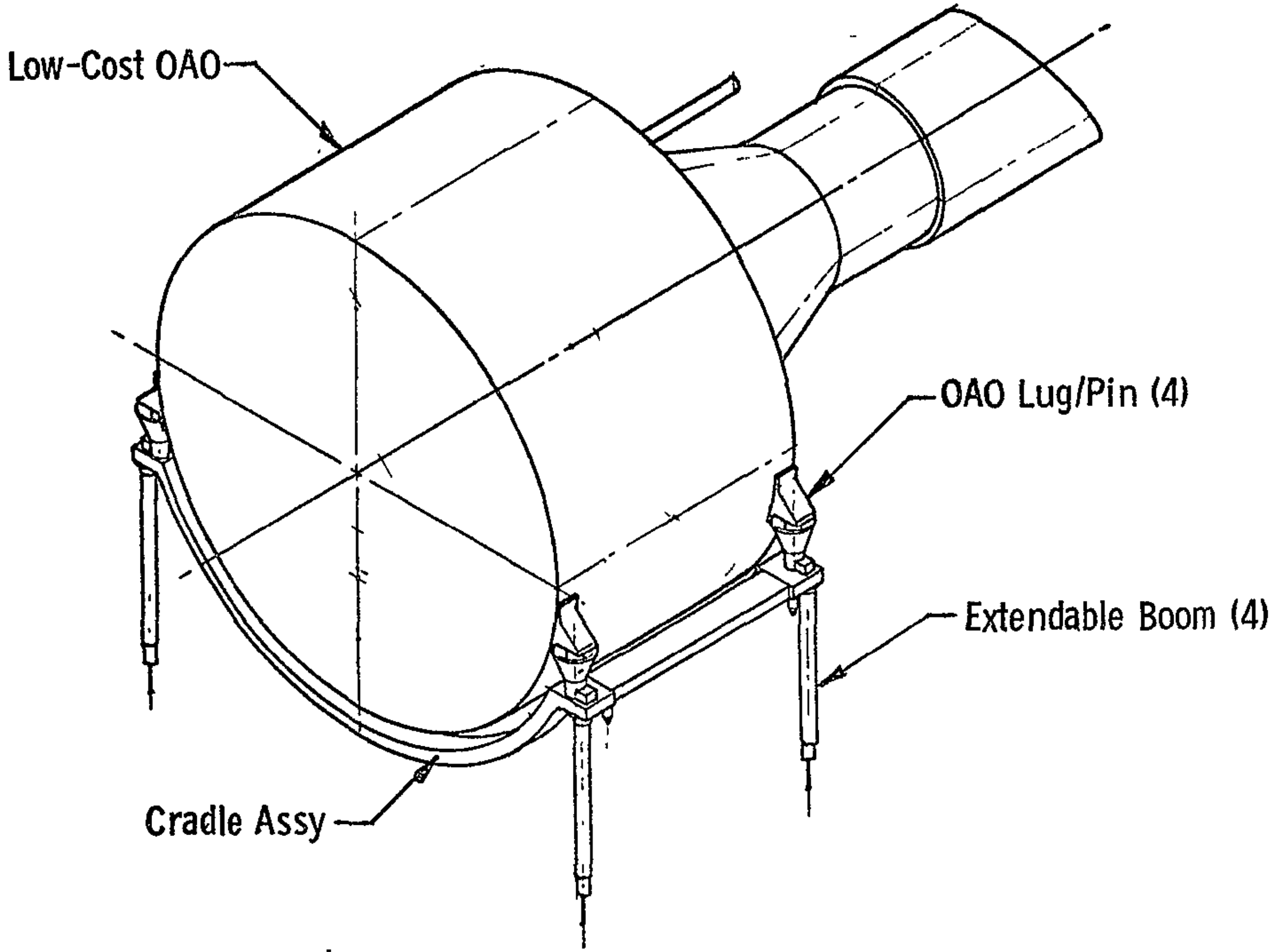


Fig. 8-9 OAO Extended Position for Deployment/Retrieval

The next phase of checkout of the OAO is initiated, using solar array power and with the OAO transmitting directly to ground to verify the final functions. When all signals are "go", the OAO is released from the cradle assembly by remote actuation of drogue funnel latches (via electrical conductor within the extendable boom).

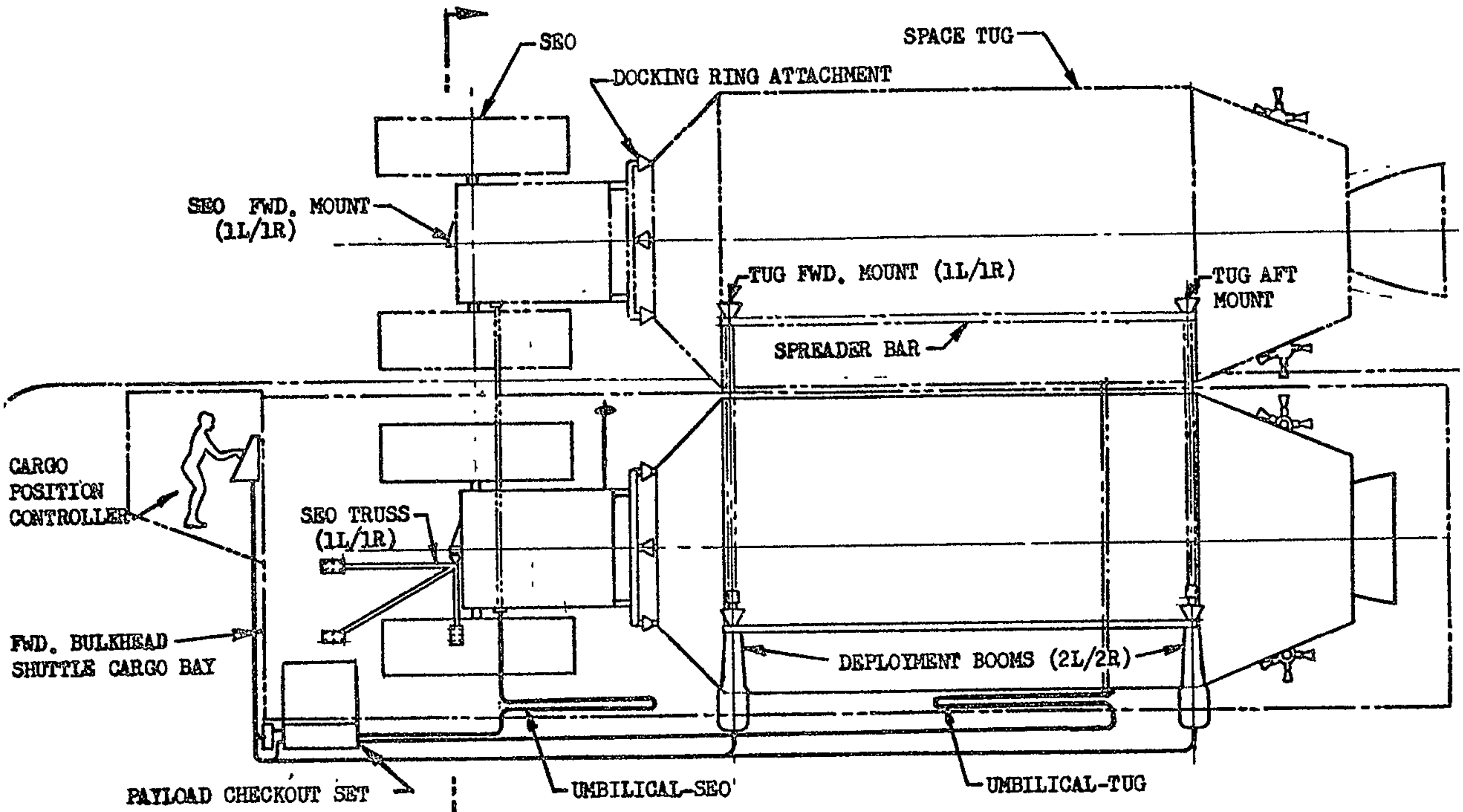
Retrieval of the OAO is performed in reverse order. The OAO is grappled from free-flight by the Tug or a telefactor robot dispatched from the Shuttle. The Tug or robot, with vernier translation and position control, will place the OAO into the aforementioned extended cradle assembly, engaging the four OAO support/hold-down pins into the drogue funnels on the cradle. The booms will then be actuated to "retract", lowering the OAO/cradle into latched position within the cargo bay.

8.3.2 Installation and Deployment/Retrieval of the Low-Cost SEO

The installation of the SEO/Tug within the Shuttle is similar to that of the OAO in that four synchronized extendable booms are used; a cradle assembly is utilized to support the Tug in the extended position. Figures 8-10a and -10b illustrate the installation arrangement. The SEO is mounted to the Tug by a docking ring attachment.

To provide lateral stabilization (vertical and sideways) of the SEO for launch/ascent and landing loads, 1L/1R support points are installed at the forward end; these supports also sustain fore-aft loads. The docking ring carries lateral loads only and is spring-mounted to the SEO to prevent loads which might be exerted by Shuttle dimensional growth or shrinkage between the fixed forward mounts on the SEO and on the Tug.

For extension of the SEO/Tug, six support latches are released; 1L/1R on the SEO and 2L/2R on the Tug. The extendable booms are then actuated to raise the SEO/Tug to an extended position (shown also in Fig. 8-11).



See Fig. 8-10b

Fig. 8-10a Shuttle Installation of SEO and Space Tug

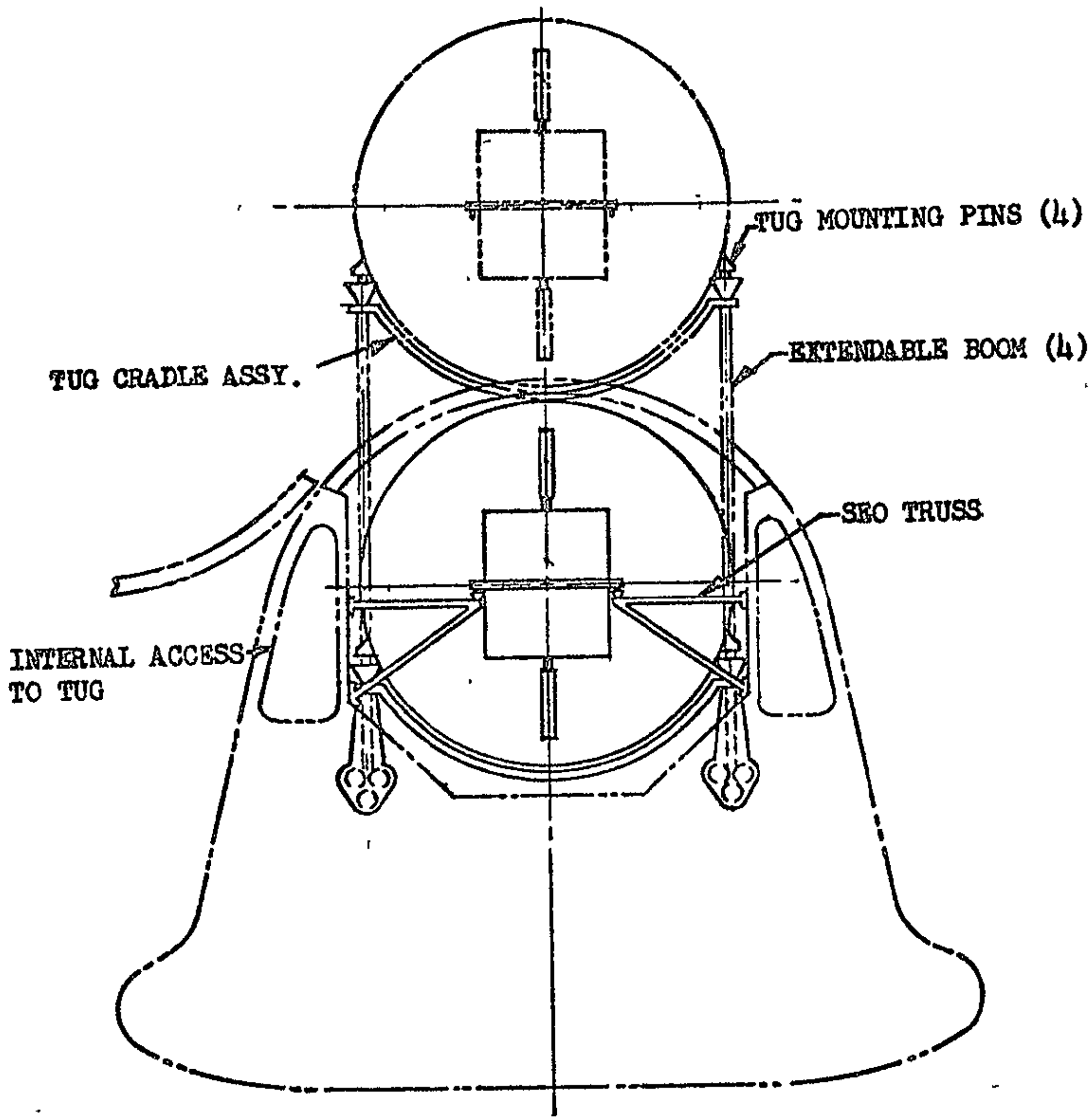


Fig. 8-10b Shuttle Installation of SEO and Space Tug

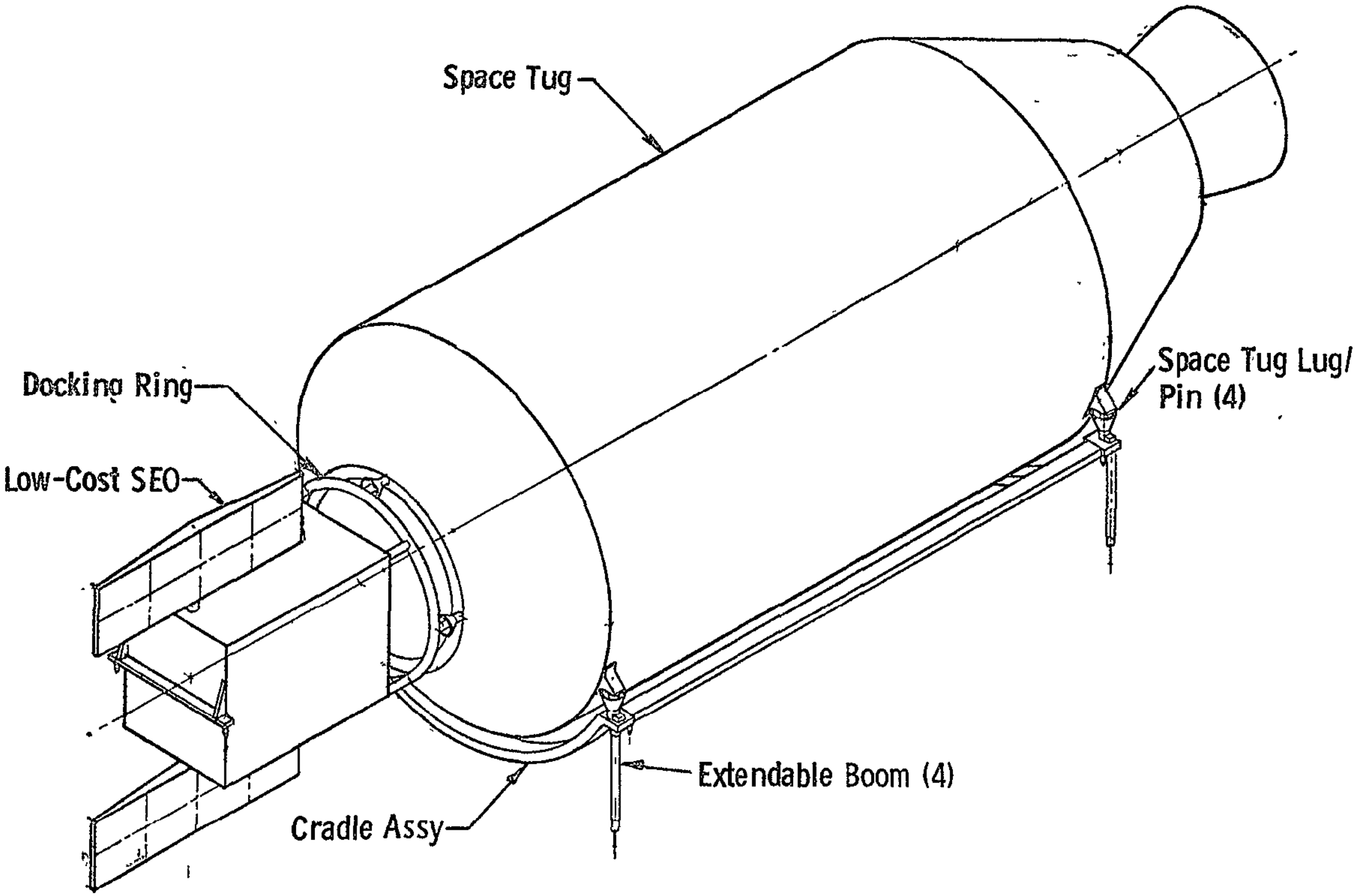


Fig. 8-11 SEO/Tug Extended Position for Deployment/Retrieval

The SEO/Tug is released from the cradle assembly (same as for OAO), the Shuttle is maneuvered to a safe station-keeping distance, and the Tug ignited by remote signal from the Shuttle. The Tug pushes the SEO to Syneq orbit position at which time the docking ring is de-coupled by remote signal from the Shuttle or from ground command. The Tug then returns to low earth orbit for rendezvous and retrieval by the Shuttle and return to earth.

Retrieval of the SEO in Syneq orbit is executed by the Tug. Equipment has been installed in the low-cost SEO which will maintain the SEO in stable mode for docking by the Tug. Four passive reflectors have been provided on the SEO docking ring face to supply transponding to rendezvous and alignment sensors and transmitters on the Tug. The remote docking in Syneq orbit has been investigated to the extent that technical feasibility has been determined; however, development of this concept, both in operational techniques and hardware, is required.

The combined SEO/Tug rendezvous with the Shuttle in low earth orbit, the Tug is mated with the extended cradle assembly, and the booms are retracted to lower the SEO/Tug into latch-down position in the Shuttle cargo bay.

8.3.3 Installation and Deployment of the Low-Cost SRS

The installation of the SRS in the Shuttle is different from the OAO and SEO in that a set of three SRSs are deployed into orbit on a single Shuttle launch. The "cradle" assembly for the SRS is therefore configured to support and deploy three payloads. Figures 8-12a and -12b illustrate the arrangement. The cradle or platform is latched down to the Shuttle cargo bay structure at four points. When these latches are released in orbit, a single extendable boom raises the platform to an extended position. The payloads are then released by a pin-puller and a set of springs pushes the SRS to a safe distance from the Shuttle where the spin rockets and the orbit positioning thruster are ignited. Only two of the three SRSs are launched. The third is carried as a spare if either of the others fail to pass predeployment checkout. The unused SRS is returned to earth for refurbishment and/or checkout and reuse.

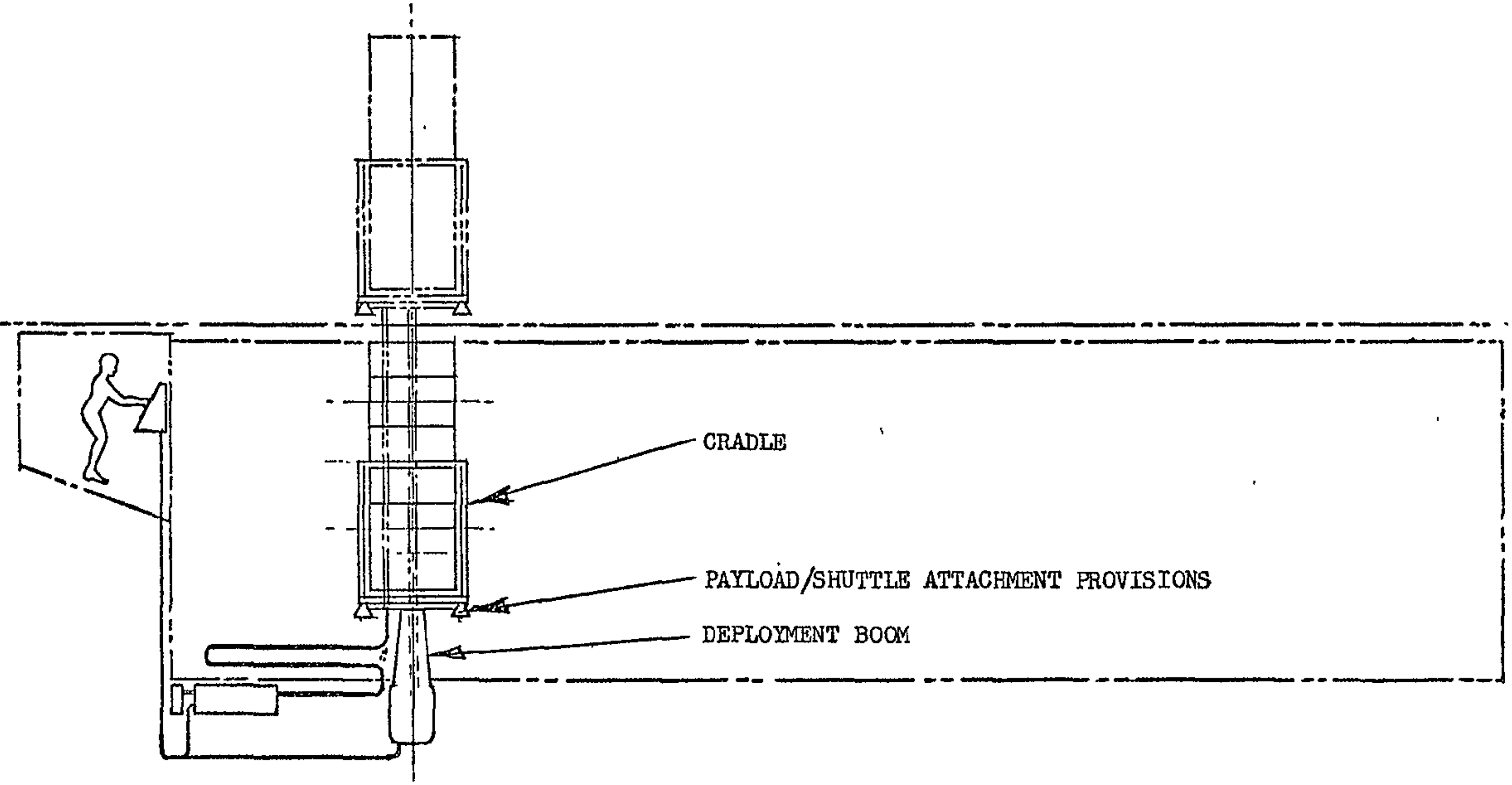


Fig. 8-12a Low-Cost SRS Space Shuttle Hold-Down/Deployment Installation

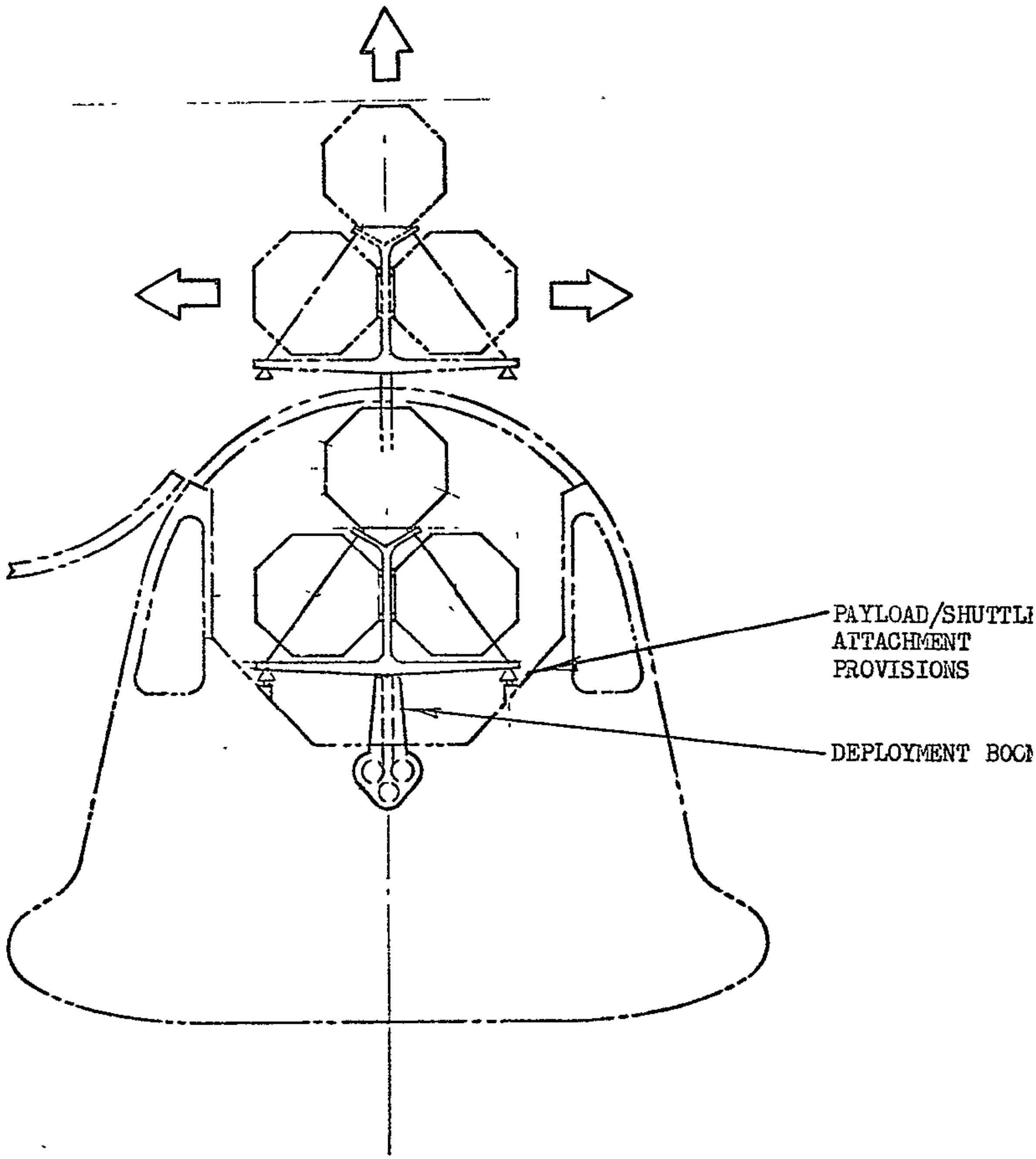


Fig. 8-12b Low-Cost SRS Hold-Down Deployment Provisions (Shuttle)

The SRS is launched as an expendable payload; retrieval is not planned.

8.3.4 The Universal Deployment/Retrieval Gear

As an aid in verifying the operational feasibility of the deployment/retrieval gear, the functional and structural characteristics were derived using supplier data on similar, but smaller, extendable booms. Weight estimates of the assembly were also made. A 3-inch diameter boom was concepted for the OAO and a 5-inch diameter boom for the SEO/Tug.

8.3.4.1 Features of the Gear. The basic concept of the extendable-boom is not new. It has been successfully used in other space applications in smaller sizes. Data from the supplier of bi-stem boom elements, SPAR Aerospace Products, Ltd., of Toronto, Canada, were utilized in developing configuration and weights. The features of deployment gear, which employs one, two, or four bi-stem motorized booms; are listed on Fig. 8-13. The design characteristics are tabulated in Fig. 8-14.

8.3.4.2 Configuration and Weight. The housing for each extendable boom was configured as shown in Fig. 8-15. The weight of a boom assembly, including sufficient metal tape to reel-out a distance of 30 ft (9.15 m), is 90 lb (40.8 kg) for a 3 in. dia. (7.62 cm dia.) boom and 302 lb (137 kg) for a 5 in. dia. (12.7 cm dia.) boom. A preliminary design of the other elements of the deployment gear was accomplished so that weight estimates could be made. The weight breakdown is listed in Fig. 8-16. The total installed weight of the gear for the OAO is 528 lb (239 kg); the weight of the gear for the SEO/Tug is 1484 lb (675 kg).

8.4 REPAIR AND REFURBISHMENT OF PAYLOADS

To provide for the maximum reduction in program cost of Shuttle-launched low-cost payloads, it is necessary to employ repair/refurbishment and reuse techniques for not only the payload but also for equipment modules and components which comprise the payload. This sub-section is devoted to describing the

- **EXTENDABLE-BOOM ASSEMBLIES**
  - Can be used singly or in sets of 2 or 4
  - Simple bolt-on mounting to cargo bay bulkhead
  - Integral electrical cable for remote latch activation
  - Sustains bending loads with Shuttle maneuvering and with payloads extended up to 40 ft above stowage position (12.2 m)
  - Rate of extention/retraction: 0.5 in per sec; 20 ft in 8 min (1.27 cm per sec; 6.1 m in 8 min.)
- **CRADLE ASSEMBLY**
  - Single welded-beam construction
  - Interchangeable end fitting assembly (4) - Drogue funnel, latch, hold-down pin
  - Varying-length spreader bars to adapt to payload length
  - Semi-circular cradles to fit payload diameter
- **SHUTTLE/SUPPORT/HOLD-DOWN**
  - Interchangeable fitting to mate with cradle - Drogue funnel, latch
  - Welded bracket or truss; bolt-attaches to stiffened attach area on cargo bay bulkheads

Fig. 8-13 Features of Universal Deployment/Retrieval Gear

TYPE: Storable Tubular Extendable Member (STEM) with two concentric spring-tube elements (BI-STEM)

TUBE ELEMENTS: Std steel spring; .020 x 8.6 in. flat, 3" dia. (30^0 gap)
.0333 x 12.1 in. flat, 5" dia. (30^0 gap)

MAXIMUM SHUTTLE ANGULAR ACCEL. (PITCH): 0.74 deg/sec^2 (or 1.3×10^{-2} rad/sec^2)

BOOM BENDING CHARACTERISTICS:**

Extension	Force at Boom Tip	Max. Shuttle-Applied Moment		Allowable Bending		Maximum Extension			
		F.S.=1	F.S.=3	3" dia.	5" dia.	F.S.=1		F.S.=3	
						3" D	5" D	3" D	5" D
10 ft	268 lb	335 ft lb	1005 ft lb	1000 ft lb	4500 ft lb	17 ft	37 ft	9 ft	22 ft
20 ft	535 lb	1338 ft lb	4012 ft lb						
30 ft	803 lb	3011 ft lb	9033 ft lb						
40 ft	1071 lb	5355 ft lb	16065 ft lb						

BOOM COMPRESSION STRENGTH: 1000 lb

ELECTRICAL POWER REQD: Retraction/Extension* - 8 watts for 3" dia; 16 watts for 5" dia

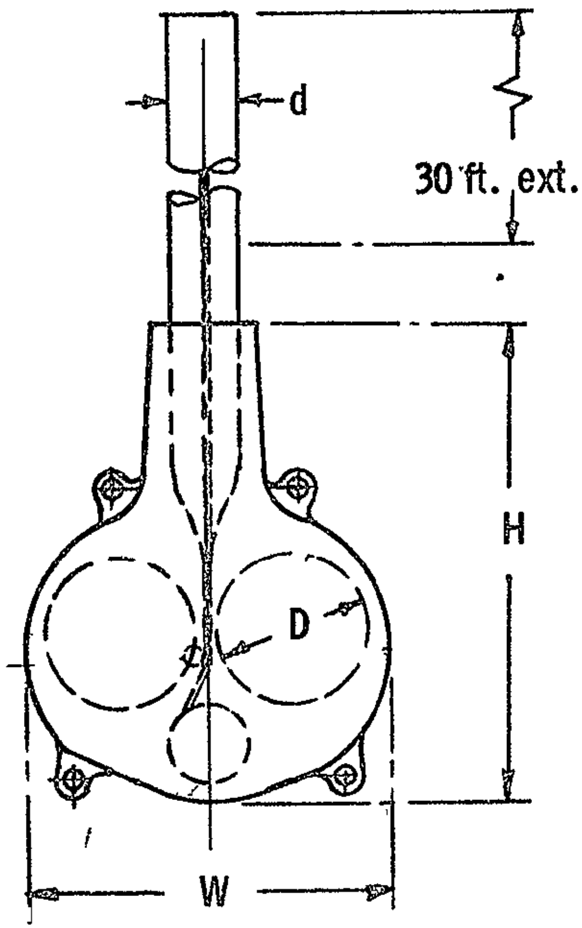
TIP DEFLECTION**

Extension	Deflection	
	3" Dia.	5" Dia.
10 ft	.004 in.	.0005 in.
20 ft	.185	.0265
30 ft	.94	.134
40 ft	3.9	.56

* Extension uses lower wattage because coiled element assists.
** SEO/Tug loads applied.

1 in. = 2.54 cm
1 ft = 0.3048 m
1 lb = 4.444 N
1 ft lb = 1.356 Nm

Fig. 8-14 Design Characteristics of Bi-Stem Deployment Booms



1 in = 2.54 cm
1 ft = 0.3048 m
1 lb = 0.4536 kg

Dimensions		
	Boom Dia.	
	3 in.	5 in.
d	3	5
D	6	10
W	13	19
H	50	70

Weights		
	Boom Dia.	
	3 in.	5 in.
Boom Elements	42 lb	122 lb
Housing Motors, etc.	48	180
Total Weight	90 lb	302 lb

Fig. 8-15 Dimensions and Weights - Extendable Boom Assy.

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HARDWARE ITEM	FOR LOW-COST OAO			FOR LOW-COST SEO & TUG		
	Qty	Unit Wt.	Total Wt.	Qty	Unit Wt.	Total Wt.
3" Extendable Boom Assy	4	90.0 lb	360.0 lb			
5" Extendable Boom Assy				4	302.0 lb	1208.0 lb
Cradle Assy.		50.0	116.0		87.0	194.0
Drogue Funnel	4	(2.0)	(8.0)	4	(2.0)	(8.0)
Latch	4	(1.0)	(4.0)	4	(1.0)	(4.0)
Cradle End Fitting	4	(3.0)	(12.0)	4	(3.0)	(12 .0)
Cradle Shear Hold-Down Pins	4	(2.0)	(8.0)	4	(2.0)	(8.0)
Cradle Beam	2	(26.0)	(52.0)	2	(28.0)	(56.0)
Spreader Bar	2	(16.0)	(32.0)	2	(53.0)	(106.0)
Supports - Cradle/Shuttle		13.0	52.0		13.0	52.0
Drogue Funnel	4	(2.0)	(8.0)	4	(2.0)	(8.0)
Latch	4	(1.0)	(4.0)	4	(1.0)	(4.0)
Truss	4	(10.0)	(40.0)	-	-	-
Bracket				4	(10.0)	(40.0)
Support - SEO/Shuttle					15.0	30.0
Drogue Funnel				2	(2.0)	(4.0)
Latch				2	(1.0)	(2.0)
Truss				2	(12.0)	(24.0)
TOTALS		153.0 lb	528.0 lb		417.0 lb	1484.0 lb

1 lb = 0.4536 kg   1 ft = 0.3048 m

Fig. 8-16 Weight Estimates - Universal Support/Deployment/Retrieval Installation for Payloads (30 ft extension)

concepts of repair and refurbishment developed during the study and illustrating the magnitude of dollar savings which can be obtained.

8.4.1 Basic Approach to Payload Repair/Refurbishment

The concept developed for repair and refurbishment includes these basic elements:

- Repair of initial-launch payload on-orbit following pre-deployment checkout
- Periodic refurbishment of the payload by replacement of equipment modules
- Orbit repair of random failures which occur between refurbishments
- Refurbishment (and reuse) of equipment modules by replacement of components therein
- Refurbishment (and reuse) of components by replacement of parts and subassemblies therein.

Figure 8-17 lists the basic steps involved in this overall concept. Throughout this discussion, the meaning of the two primary terms are:

"Repair" - the replacement of a failed module with a new or refurbished module thereby returning the payload to operational status (but not changing the overall life expectancy of the payload).

"Refurbishment" - the replacement of all hardware elements which have degraded over the operating time, thereby restoring the payload (or module or component) to its initial operating life expectancy.

- REPAIR ON ORBIT
  - Carry as spares on initial launch: 10 different modules (total wt. 2093 lb) - OAO; 11 different modules (total wt. 1423 lb) - SEO
  - Checkout of payload on orbit
  - Replace any module which has failed or degraded in launch/ascent
  - Return failed module for ground refurbishment
- REFURBISHMENT OF PAYLOAD
  - Periodically replace the orbiting payload with a refurbished (at nom. 1-yr. interval for OAO; at 2-yr. intervals for SEO)
  - Retrieve the "used" payload from orbit with Shuttle or Tug/Shuttle and return to earth.
  - Remove used/failed modules from space frame
  - Install new (or refurbished) modules
  - Perform system-level payload checkout
- REFURBISHMENT OF MODULES
  - Remove module cover and equipment components
  - Install new (or refurbished) components into module
  - Test module in spacecraft simulator, using standard checkout set
- REFURBISHMENT OF COMPONENTS

LOW-COST PAYLOAD REPAIR/REFURBISHMENT APPROACH

Fig. 8-17

8.4.2 Analysis of On-Orbit Module Replacement

To assure that the concept of replacement of equipment modules in payloads on orbit was sound, feasibility analyses were made of the areas affecting the actual hardware implementation of the concept. Figure 8-18 lists these analyses. The concept was determined to be feasible and compatible with the Shuttle operational modes.

8.4.3 Repair of Payloads on Orbit

Repair on orbit of payloads falls into two categories: (1) repair following the rigorous launch/ascent and (2) repair of random failures which may occur between the periodic refurbishment points.

8.4.3.1 Checkout and Repair After Launch/Ascent. Investigation of payload failure modes has revealed that a significant percentage of total failures occurs in the launch/ascent phase of payload total life or shortly thereafter. Although further detailed analysis of this condition is required, it appears desirable to isolate and eliminate these failures where possible. It is, therefore, proposed that replacement (spare) modules be carried to orbit with the payload. Checkout of the payload on orbit (using the Shuttle-carried payload checkout set) will reveal any failure or degraded performance of a module; the module can then be replaced prior to payload deployment to orbit.

There are eleven different modules in each payload with multiple quantities of some of these. It is planned (at least for the early launches, until statistical failure data can be accumulated) that one of each of the different modules (except the solar array unit) be carried to orbit with each payload launch The list of modules for the OAO is shown on Fig. 8-19 . One set (excluding the solar array) weighs 2093 lb (950 kg).

Equivalent data for the SEO modules are given on Fig. 8-20. The set of spares for the SEO (excluding the solar array paddle) weighs 1354 lb (613 kg).

- DEACTIVATION AND ATTITUDE STABILIZATION OF PAYLOAD
- EVA VS REMOTE MANIPULATORS
- MANNED AND UNMANNED SPACE TUG OPERATIONS WITH THE SHUTTLE
- RECALIBRATION OF PAYLOAD AFTER MODULE REPLACEMENT
- GROUND-BASE RESPONSE TIME
- TURN AROUND TIME FOR GROUND REFURBISHMENT
- ON-ORBIT RE-CHECKOUT AFTER MAINTENANCE/REFURBISHMENT
- PAYLOAD WEAROUT/RELIABILITY VS REFURBISHMENT CYCLE

IN-ORBIT MAINTENANCE/REFURBISHMENT ANALYSES

Fig. 8-18

Subsystem	Module No.	Module Type	Quantity per Payload	Delivered Cost per Single Module ($ Million)	Unit Weight (lb)
Electrical	1	Battery	1	.919	448
	2	Power	1	.434	196
	-	Solar Array	2 ea.	.368	175
CDPI	1	Computer	1	.716	322
	2	Communication	1	.904	104
Stabilization & Control	1,2	Primary Attitude	1 ea.	1.011	161
	3	Secondary Att.	1	1.055	150
	4	Wheel	1	.815	186
Attitude Control	1,2 3,4	Attitude Control	1 ea.	.241	207
Experiment	1	Electronic	1	.765	156
	2	Electro-Mech.	1	.917	173
		TOTALS		8.145	2268

1 lb = 0.4536 kg

LOW-COST OAO MODULES

Fig. 8-19

Subsystem	Module No.	Module Type	Qty. per P/L	Delivered Cost Per Single Module ($1000)	Unit Weight (lb)
Electrical	1	Battery	1	187	170
	2	Power Control	1	402	106
	3,4	Paddle Drive	1 ea	120	39
	5,6	Solar Paddle	1 ea	264	69
Commun., Data Proc., Instrumen.	1,2	Data Handling	1 ea	665	81
	3	Communications	1	889	87
Stabilization & Control	1	Sensing & Flight Control	1	1117	98
	2	Momentum	1	504	125
Attitude Control	1,2 3,4	Attitude Control	1 ea	139	143
Experiment	1	Photographic	1	1293	419
	2	Vidicon Camera	1	381	86
			Totals	5961	1423

1 lb = 0.4536 kg

LOW-COST SEO MODULES AND COSTS

Fig. 8-20

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8.4.3.2 Repair of Random Failures. Because of the comparatively high cost of transportation to perform a repair revisit, it is not proposed to perform repair only on a low earth orbit payload (OAO) when a random catastrophic failure occurs within the last three months of a 1-year operating period. Rather, a refurbishment will be performed. If a failure occurs prior to the nominally-scheduled 1-year point, the refurbishment will be rescheduled. Similarly, if satisfactory operation of the OAO continues beyond the 1-year point, the refurbishment can be delayed. The statistical probability of failure occurrence (or wearout) has been assumed to average a 1 year time period for the OAO.

On a Syneq orbit payload (SEO), it is not proposed to perform repair only when a random catastrophic failure occurs. Rather a payload replacement will be performed and the failed SEO will be returned to earth for refurbishment. The statistical probability of occurrence of random failures on the SEO has been assumed to average a 2-year time period.

8.4.3.3 Spares for On-Orbit Repair. The cost of spares replacements for SEO repair has been estimated. It has been assumed that a group of 4 SEO launches will require replacement of two modules or an average of one module for each 2 SEOs. The average cost of a module is $542,000 (total of 11-module set divided by the quantity). The average cost is assumed to be maintained in a randomly distributed fashion over the program period (detail analysis, proposed for follow-on study, will be able to ascertain module failures and their occurrence probability). This average module cost divided by the recurring cost of two SEOs ($19.66 million) derives an average repair cost totalling 2.75 percent of the SEO recurring cost. Multiplying the "average" repair cost of $271,000 per launch by the 20 launches required for the 10-year program derives $5.42 million required for on-orbit repair. These costs have been included in the allotment for total spares in the cost estimates for the low-cost SEO.

Similarly, the average cost of a module for the OAO is $740,000, which is also assumed to be applied randomly over a 6-year program with one module requiring

replacement each year. The total repair cost for the 6-year program would be $4.45 million. Costs to cover these repairs are included in the conservative estimate for all program spares of $18.66 million for the low-cost OAO.

8.4.4 Refurbishment of Payloads

8.4.4.1 OAO Refurbishment in Orbit. It is planned to periodically (at the end of each 1-year operating period, or as varied by actual failure experience) launch a Shuttle with a set of replacement modules. The OAO would be retrieved by the Shuttle in low earth orbit and all modules replaced while the OAO was tethered to the Shuttle on the extended deployment/retrieval gear or within the Shuttle cargo bay. The new modules would be installed in the OAO: (1) by Shuttle crew members working in EVA or non-pressurized IVA; (2) by tele-factor robots remotely controlled from the Shuttle crew compartment; (3) by automated devices; or (4) by combinations of these. Figure 8-21 shows pictorially four of these modes.

The Shuttle will return the failed or spent modules to earth for refurbishment.

8.4.4.2 SEO Refurbishment. It is planned to periodically (at the end of each 2-year operating period, or as varied by actual failure experience) launch a replacement SEO with a Shuttle/Tug. The failed or spent SEO would be returned to earth for refurbishment. The low-cost SEO will accommodate refurbishment on orbit but because the Tug must be returned to earth for propellant refill in the mode assumed, the SEO is returned to earth for the refurbishment. For this operation, a ready-to-launch replacement SEO must be available on the ground at all times.

The first, second, and third of a set of 4 SEOs must be refurbished and ready for relaunch within 60 days (to match the average launch cycle time of 60 days between launches). Refurbishment of the SEO will comprise replacement of the equipment modules and re-checkout of each SEO. Because of the longer time span

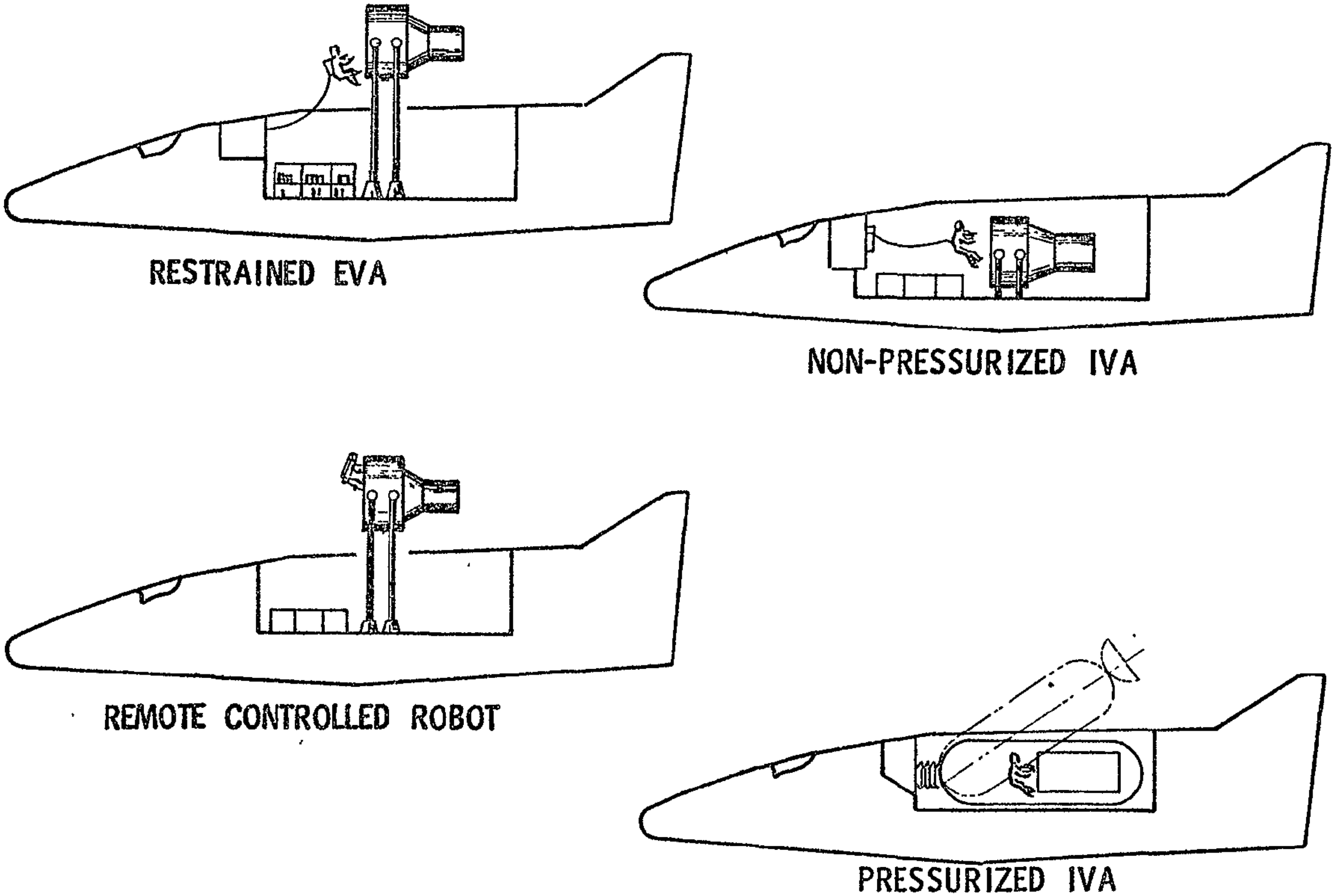

Fig. 8-21 In-Orbit Repair/Refurbishment Concepts

required for module-level refurbishment, two sets of modules should be held in standby; this allows approximately 4 months for turnaround time for refurbishment of a set of modules.

8.4.5 Refurbishment of Components

8.4.5.1 Component Refurbishment Concept. The lowest level of refurbishment planned is the component (two or more components are installed in each equipment module; Section 5 of this report provides a detail description of each module of the OAO and SEO and the components included therein). It is planned that component refurbishment will be accomplished in field repair/refurbishment depots which are manned by technicians skilled in the various separate subsystems of several payloads; for example, electrical components, stabilization and control components, etc. Parts for this component refurbishment presumably would have been ordered at the time of payload initial procurement (following a thorough logistic analysis and spares provisioning) and would be stocked in the depot in bonded stores. Also, required quantities of test sets would be supplied to the depot to allow complete test/checkout of the component after refurbishment.

The actual refurbishment of the component should be limited to replacement of those parts and sub-assemblies which can be readily removed and installed; conversely reuse without replacement is planned for as many pieces of hardware as possible. Contractor-supplied repair and overhaul instruction manuals would be procured and used by depot personnel as guides for the refurbishment work. To facilitate the refurbishment activity, the design of the component hardware should be carefully done to be compatible with the basic depot dis-assembly - assembly procedures.

8.4.5.2 Refurbishment of Low-Cost OAO Components. An analysis has been made of the refurbishability of the various components in the low-cost OAO modules. Figures 8-22a, -22b, and -22c list: (1) the components; (2) the assumed replaceable hardware elements; and (3) the estimated cost of replacement parts,

Subsystem/ Module	Component	Replaceable Hardware Elements	Comp. Initial Recur. Cost	Cost of Component Refurbish.				Comp. Saving
				Repl. Parts	Disassy Assy	QA & Test	Total	
ELECTRICAL:								
Battery (No. 1)	Battery (3)	All	.030	-	-	-	-	-
	Pow.Cont.Unit	PCBs	.240	.170	.004	.006	.180	.060
	Pow.Reg.Unit	Xfmr, PCBs	.100	.070	.002	.003	.075	.025
	St. of Chg.Un	PCBs	.080	.050	.001	.001	.052	.028
	Diode Box	PCBs	.040	.030	.001	.001	.032	.008
	Grd.Pow.Rel.	Cont.Points	.080	.001	.001	.001	.003	.005
	Batt.Current Shunt	Instrument.	.013	.007	.001	.001	.009	.004
Power (No. 2)	Volt.Reg/Conv.	PCBs	.060	.040	.002	.002	.044	.016
	Volt.Inverter	Xfmr, PCBs	.090	.060	.002	.003	.065	.025
	Pow.Dist.Unit	Fuses, CBs	.050	.030	.005	.002	.037	.013
CDPI:								
Computer (No. 1)	Computer (2)	All	.110	-	-	-	-	-
	I/F Tim.Unit(2)	PCBs	.125	.080	.004	.005	.089	.036
	Cold Plate	None	.010	-	-	-	-	.010
Comm. (No. 2)	Tape Recorder	PCBs,Motors	.125	.080	.006	.004	.090	.035
	NB Xmtr	PCBs	.030	.020	.001	.001	.022	.008
	WB Xmtr	PCBs	.150	.100	.003	.004	.107	.043
	Cmd. Rcvr(2)	PCBs	.080	.050	.002	.002	.054	.026

COMPONENT LEVEL REFURBISHMENT COSTS - 040 ($ Million) (Sheet 1 of 3)

Subsystem/ Module	Component	Replaceable Hardware Elements	Comp. Initial Recur. Cost	Cost of Component Refurbish.				Comp. Saving
				Repl. Parts	Disassy Assy	QA & Test	Total	
S & C: Primary Att. Ref. (No. 1, 2)	Gimb. Star Tracker	Electronics	.300	.180	.010	.005	.195	.105
	GST Electronics	PCBs	.100	.060	.004	.004	.068	.032
	Inert. Ref. Unit	Gyros, PCBs	.185	.150	.007	.005	.162	.023
Sec. Att. Ref. (No. 3)	Gimb. Star Tracker	Actuators, Mech.	.300	.180	.010	.005	.195	.105
	GST Electron.	PCBs	.100	.060	.004	.004	.068	.032
	BS Star Track.	Mech.	.165	.080	.008	.003	.091	.074
	BST Electron.	PCBs	.041	.025	.002	.002	.029	.012
	SAS Electron.	PCBs	.020	.010	.001	.001	.012	.008
Wheel (No. 4)	Moment Wheels(3)	Bearings, Mech, Motors	.300	.150	.010	.005	.165	.135
	Wheel Elect.	PCBs	.050	.030	.002	.003	.035	.015
	Fine Solar Asp. Sensor	-	.015	-	-	-	-	-
	FSAS Elect.	PCBs	.029	.018	.001	.001	.020	.009

COMPONENT LEVEL REFURBISHMENT COSTS - OAO (Sheet 2 of 3)
($ Million)

Fig. 8-22b

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Subsystem/ Module	Component	Replaceable Hardware Elements	Comp. Initial Recur. Cost	Cost of Component Refurb.				Comp. Saving
				Repl. Parts	Disassy Assy	QA & Test	Total	
ATTITUDE CONTROL:	Tank	(Reuse)	.005	-	-	-	-	.005
Attitude Control No. 1, 2, 3, 4)	Valves	Seats, Solenoids, Seals	.011	.002	.004	.002	.008	.003
	Regulators	Springs,	.027	.004	.004	.002	.010	.017
	Nozzles	Seats (Reuse)	.012	-	-	-	-	.012
	Transducers	All	.002	-	-	-	-	-
	Solar Asp. Sens	All	.004	-	-	-	-	-
	Thruster Elect	PCBs	.017	.010	.001	.001	.012	.005
EXPERIMENT:								
Exper. Electronics	Electronic Unit	PCBs	.680	.400	.010	.010	.420	.260
Electro-Mech.	Detector	All	.079	-	-	-	-	-
	Focal Plane Assy	Bearings, etc	.170	.040	.010	.005	.055	.115
	Fine Guid Assy	PCBs	.390	.280	.020	.010	.310	.080

COMPONENT LEVEL REFURBISHMENT COSTS - OAO (Sheet 3 of 3)
($ Million)

Fig. 8-22c

disassembly/assembly labor, QA and test labor. In some instances, the component is not refurbishable; an example is the battery in the electrical power subsystem. In other cases, a new component is relatively inexpensive when compared to the replacement parts and labor needed to refurbish it; in these cases refurbishment is not planned -- examples are the computer, solar aspect sensor, transducers.

8.4.5.3 Refurbishment of Low-Cost SEO Components. A similar analysis was made of the refurbishability of the low-cost SEO components. Figures 8-23a, -23b, and -23c list data equivalent to that described in 8.4.5.2 for the OAO.

8.4.6 Refurbishment of Modules

8.4.6.1 Basic Approach. The basic approach to module refurbishment is listed in Fig. 8-24. The modules have been designed for this type of easy disassembly and reassembly (see description of typical modules in Section 5). The modules which are not proposed candidates for refurbishment are:

- Solar Array Paddle (OAO and SEO)

  The principal costs are embodied in the solar-cell panels. Although these are designed to be readily replaceable the field replacement of all of these panels involves a considerable amount of disassembly, assembly, and test labor. It therefore appears desirable to order the complete solar array paddle as a replacement article.

- TV Camera (OAO)

  This item is a highly sophisticated piece of hardware involving a combination of electronic scanning devices. Although redesign for refurb may be feasible, it was not considered during this study.

8.4.6.2 OAO Module Refurbishment. The costs of refurbishment for the low-cost OAO modules have been estimated and are tabulated in Fig. 8-25. The "replacement

Subsystem/ Module	Component	Replaceable Hardware Elements	Comp. Initial Recur. Cost	Cost of Component Refurbish.				Comp Saving
				Repl. Parts	Disassy Assy	QA & Test	Total	
<u>Electrical</u>:								
Battery (No. 1)	Battery (4)	All	.040	-	-	-	-	-
	Current Shunt	Instrument.	.013	.007	.001	.001	.009	.004
Power Control (No. 2)	Bat. Chg.Cont.	PCBs						
	St.of Chg.Unit	PCBs						
	Pow.Dist. Unit	Fuses, CBs	.163	.123	.010	.010	.143	.020
	Motor Cont-S/A	PCBs,Switches						
	Pulse Gen-S/A	PCBs						
	Grnd.Pow.Relay	Cont.Points	.008	.001	.001	.001	.003	.005
Paddle Drive No. 3,4	Drive Motor(2) Gear Box	All (Reuse, Lube)	.028	.012	.001	.001	.014	.014

<u>COMPONENT-LEVEL REFURBISHMENT COSTS - SEO</u> (Sheet 1 of 3)

($ Million)

Fig. 8-23a

Subsystem/ Module	Component	Replaceable Hardware Elements	Comp Initial Recur. Cost	Cost of Component Refurbish.				Comp Saving
				Repl. Parts	Disassy Assy	QA & Test	Total	
CDPI:	Tape Recorder	PCBs, Motors	.250	.170	.010	.004	.184	.066
Data Hdlg.	PCM Mul/Enc.	PCBs	.085	.050	.001	.002	.053	.032
	Cmd. Decoder	PCBs	.090	.060	.001	.002	.063	.027
Commun.	Transponder(2)	PCBs	.360	.216	.003	.004	.223	.137
	TWTA	PCBs, TWT	.075	.060	.002	.002	.064	.011
	Mod. Selector	PCBs	.090	.060	.001	.001	.062	.028
S & C:	Flight Control Electronics	PCBs	.592	.350	.005	.010	.365	.227
Sensing & Flt. Control	Horiz. Sens. (2)	Electronics	.150	.100	.004	.005	.109	.041
	Solar Aspect Sensor(2)	All	.003	-	-	-	-	-
Momentum	Gimb. Wheel	Bearings, Mech; Motors	.191	.100	.010	.005	.115	.076
	Wh. Drive Elec.	All	.004	-	-	-	-	-
	Rate Gyro	All	.025	-	-	-	-	-
	Wh. Safety Shld	None	.002	-	-	-	-	.002

COMPONENT-LEVEL REFURBISHMENT COSTS - SEO (Sheet 2 of 3)

($ Million)

Fig. 8-23b

Subsystem/ Module	Component	Replaceable Hardware Elements	Comp. Initial Recur. Cost	Cost of Component Refurbish.				Comp Saving
				Repl Parts	Disassy Assy	QA & Test	Total	
Attitude Control:	Tank	(Reuse)	.005	-	-	-	-	.005
	Valves	Seats, Solenoids, Seals	.012	.002	.004	.002	.008	.004
Attitude Control	Press. Reg.	Spring, Seats	.015	.003	.003	.001	.007	.008
	Valve Clust./ Nozzles	Valve Cluster	.006	.002	.001	.001	.004	.002
	Plumbing	Assy. Braze	.003	-	.002	.001	.003	-
Experiment:	Camera	Seals, Mech.	.043	.015	.005	.002	.022	.021
Photographic	Proces./Dryer	Mech.	.012	.003	.001	.001	.005	.008
	Opt/Mech. Scan.	Electron.	.057	.030	.002	.005	.037	.020
	Film Hdlg. Assy	Vac. Rate, Motors	.060	.010	.004	.002	.016	.044
	Electron. Assy.	PCBs	.098	.066	.004	.003	.073	.025
	N_2 Supply	None	.005	-	-	-	-	.005

COMPONENT-LEVEL REFURBISHMENT COSTS - SEO (Sheet 3 of 3)

($ Million)

Fig. 8-23c

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- DISASSEMBLY - Remove module cover, remove components
- HARDWARE REPLACEMENT - Replace components with new or refurbished items. Replace parts in selected components. Replacement hardware procured as spares and delivered concurrently with payload.
- COMPONENT TEST - Test all new or refurbished components prior to installation into module using checkout set.
- INSPECTION - Inspect all hardware to be re-used in accordance with payload maintenance manual (includes mechanical parts, structural elements, chassis, electrical cabling, etc.)
- ADJUSTMENT/ALIGNMENT - Check and adjust or calibrate as necessary in accordance with payload maintenance manual.
- ASSEMBLY AND TEST - Reassemble module and perform complete functional checkout using checkout set and payload simulator.

GROUND REFURBISHMENT APPROACH FOR MODULES
(FIELD REPAIR DEPOT)

Fig. 8-24

	Repl. Parts	Eng. Sup. 5%	Comp. Disassy Assy.	Module Disassy Assy	Comp. QA & Test	Module QA & Test (80 %)	Total Refurb Cost	% of New Cost
Electrical - Battery No. 1	.358	.008	.010	.040	.013	.136	.565	62
Power No. 2	.130	.004	.009	.022	.007	.072	.244	57
Solar Array	-	-	-	-	-	-	-	100
CDPI - Computer No. 1	.190	.010	.004	.019	.005	.144	.372	52
Comm. No. 2	.250	.010	.012	.019	.011	.144	.446	49
S&C - Prim. Att. No. 1, 2	.390	.007	.021	.017	.014	.120	.569	56
Sec. Att. No. 3	.355	.007	.025	.017	.015	.120	.539	51
Wheel No. 4	.213	.007	.013	.017	.009	.120	.379	47
ACS - No. 1, 2, 3, 4	.022	.002	.009	.018	.005	.032	.088	22
Exper. - Electron No. 1	.400	.005	.010	.012	.010	.048	.485	64
Elect.-Mech No. 2	.399	.008	.030	.020	.015	.096	.568	62

COST OF MODULE/COMPONENT REFURBISHMENT - OAO

Fig. 8-25

parts" cost includes both the cost of parts required to refurbish the components and the cost of new replacement components (non-refurbishable). A nominal amount of support engineering has been included for the field depot work (5 percent of the engineering used by the contractor in the initial fab, assembly, and test of the delivered OAO hardware). Disassembly/assembly labor and QA/test labor costs are shown separately for the component and module-level work. The cost of a refurbished module ranges between 22 percent and 64 percent of the cost of a new module.

8.4.6.3 SEO Module Refurbishment. Equivalent data for the SEO modules are tabulated in Fig. 8-26. The cost of refurbished modules range from 39 percent to 66 percent the cost of a new module.

8.4.7 Cost Savings with Refurbishment

Total savings on a typical unmanned payload program are quite significant. In the following paragraphs, the module replacement cycle is explained and the program dollar savings are provided for typical OAO and SEO payloads. Similar savings can accrue on other programs wherein the refurbishment and reuse principle is employed.

8.4.7.1 Module Replacement Schedule and Costs for 6-Year OAO Program. The module replacement schedule (and module replacement costs) are tabulated in Fig. 8-27. All modules are replaced at 1-year intervals except the solar array and the attitude control modules. The solar arrays have been designed to operate for a 3-year period and are replaced once at the end of the third year. The attitude control modules, with inherently high reliability are also designed to operate for 3 years. To obtain the maximum redundant usage, two of the four modules are replaced at the end of each 2-year period. The summary indicates that an OAO refurbishment costs $4.91 million (or $5.65 million, at the end of the third year) compared to a new OAO costing $15.81 million. The cost of installing the replacement modules and re-checkout of the OAO is nominally low and has been left out of the summary figures.

	Repl. Parts	Eng. Sup. 5%	Comp Disassy Assy	Module Disassy Assy	Comp. QA & Test	Module QA & Test (80%)	Total Refurb Cost	% of New Cost
Electrical - Battery No. 1	.047	.006	.001	.002	.001	.060	.117	63
Power Cont. No. 2	.124	.012	.011	.006	.011	.100	.264	66
(ea) Paddle Dr. No. 3, 4	.012	.006	.001	.003	.001	.032	.055	46
(ea) Solar Paddle No 5, 6	-	-	-	-	-	-	-	100
CDPI - Data Handling, (ea) No. 1, 2	.280	.014	.012	.003	.008	.056	.373	56
Commun. No. 3	.336	.012	.006	.002	.007	.080	.443	50
S&C - Flight Cont. No. 1	.453	.025	.009	.006	.015	.108	.616	55
Momentum No. 2	.129	.020	.010	.006	.005	.072	.242	48
Attitude Control (ea) No. 1, 2, 3, 4	.007	.003	.010	.003	.005	.032	.060	43
Exper. - Photo No. 1	.124	.080	.016	.016	.013	.252	.501	39
TV Camera No. 2	-	-	-	-	-	-	-	100

COST OF MODULE/COMPONENT REFURBISHMENT - SEO

Fig. 8-26

Subsystem/Module	MODULES REPLACED & COST ($ Million)				
	End of 1st Year	End of 2nd Year	End of 3rd Year	End of 4th Year	End of 5th Year
Electrical - Battery No. 1	.565	.565	.565	.565	.565
Power No. 2	.244	.244	.244	.244	.244
Solar Array (2)	-	-	.736	-	-
CDPI - Computer No. 1	.372	.372	.372	.372	.372
Commun. No. 2	.446	.446	.446	.446	.446
S & C - Prim. Att. No. 1	.569	.569	.569	.569	.569
No. 2	.569	.569	.569	.569	.569
Sec. Att. No. 3	.539	.539	.539	.539	.539
Wheel No. 4	.379	.379	.379	.379	.379
Attitude Control No. 1	.088	-	.088	-	.088
No. 2	.088	-	.088	-	.088
No. 3	-	.088	-	.088	-
No. 4	-	.088	-	.088	-
Experiment - Electron No. 1	.485	.485	.485	.485	.485
Elec. Mech. No. 2	.568	.568	.568	.568	.568
Refurbished OAO	$\boxed{4.91}$	4.91	$\boxed{5.65}$	4.91	4.91
New OAO (Ref.)	15.81	15.81	15.81	15.81	15.81

TYPICAL MODULE REPLACEMENT SCHEDULE AND COST OF REFURBISHMENT FOR LOW-COST OAO
(6-Year Mission)

Fig. 8-27

8.4.7.2 Module Replacement Schedule and Costs for 10-Year SEO Program. Similar data are provided on Fig. 8-28 for the SEO. The SEO modules are replaced at 2-year intervals. The refurbished SEO costs $3.54 million (or $4.07 million at the end of the 4th and 8th years) as compared with a new SEO cost of $9.83 million.

8.4.7.3 Expendable vs Refurbished Hardware Costs. A summary was made of the total program costs for hardware, comparing an expendable low-cost payload with a refurbished low-cost payload (the latter is Shuttle-launched and supported). Figure 8-29 is a tabulation of these data. For a 6-year OAO program, payload hardware required for the refurbished approach is only 61 percent of equivalent expendable hardware cost. For a 10-year SEO program, the refurbished approach costs only 51 percent of the expendable payload approach. Further savings are discussed in par. 8.4.7.4.

8.4.7.4 Overall Program Costs with Refurbishment. Estimates for complete programs, including launch and operations costs, have been estimated, utilizing the basic cost data derived in the bottom-up cost estimates on the low-cost payloads (see Section 6 for detail cost data). The summary of cost savings for the low-cost OAO, cumulative by year for a 6-year period, are tabulated in Fig. 8-30. The totals show a saving of $178.09 million (about 50 percent) using a Shuttle-supported refurbished-payload approach in lieu of expendable low-cost OAOs launched on low-cost expendable launch vehicles.

The equivalent data for a 10-year SEO program, with cumulative figures for 4, 6, 8, and 10 years; are tabulated in Fig. 8-31. Here again, the savings of $360.87 million for the 10-year program represent a saving of about 50 percent for the Shuttle-supported refurbished SEO approach.

8.4.7.5 Refurbishment Ratio. So that Aerospace Corporation could apply the refurbishment principal to the Mission Model, a general refurbishment ratio was established for the OAO and SEO low-cost payloads:

Subsystem	MODULES REPLACED AND COST ($ Million)							
	End of 2nd Yr.		End of 4th Yr.		End of 6th Yr.		End of 8th Yr.	
	Mod. No.	$	Mod. No	$	Mod. No	$	Mod. No.	$
Electrical	1 2 3 4	.117 .264 .055 .055	1 2 3 4 5 6	.117 .264 .055 .055 .264 .264	1 2 3 4	.117 .264 .055 .055	1 2 3 4 5 6	.117 .264 .055 .055 .264 .264
Communications, Data Processing, Instrumentation	1 2 3	.373 .373 .443	1 2 3	.373 .373 .443	1 2 3	.373 .373 .443	1 2 3	.373 .373 .443
Stabilization & Control	1 2	.616 .242	1 2	.616 .242	1 2	.616 .242	1 2	.616 .242
Attitude Control	1 2	.060 .060	3 4	.060 .060	1 2	.060 .060	3 4	.060 .060
Experiment	1 2	.501 .381	1 2	.501 .381	1 2	.501 .381	1 2	.501 .381
Refurbished SEO		3.54		4.07		3.54		4.07
New SEO (Refurb.)		9.83		9.83		9.83		9.83

TYPICAL MODULE REPLACEMENT SCHEDULE AND COST OF REFURBISHMENT FOR LOW-COST SEO
(10-Year Mission)

Fig. 8-28

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Cost Element	6-Year OAO Program		10-Year SEO Program	
	Expendable	Refurbished	Expendable	Refurbished
Non-Recurring	$ 89.41	$ 84.03	$ 97.99	$ 85.70
Unit Cost - Delivered Payloads	114.84	15.81	234.56	49.15
Replacement Modules - New	-	0.74	-	2.48
Hardware for Module/ Component Refurb	-	15.60	-	28.96
Hardware Totals	$ 204.25	$ 116.18	$ 332.55	$ 166.29
Refurb. Costs	-	8.96	-	29.44
Program Totals	$ 204.25	$ 125.14	$ 332.55	$ 195.73

Fig. 8-29 Expendable vs Refurbished Low-Cost Payload Costs ($ Million)

COST ELEMENTS	EXPENDABLE ($ MILLION)						SPACE SHUTTLE ($ MILLION)					
	Length of Program						Length of Program					
	1 Yr.	2 Yr.	3 Yr.	4 Yr.	5 Yr.	6 Yr.	1 Yr.	2 Yr.	3 Yr.	4 Yr.	5 Yr.	6 Yr.
Non-Recurring Cost	89.41	89.41	89.41	89.41	89.41	89.41	84.03	84.03	84.03	84.03	84.03	84.03
Unit Cost	19.14	38.28	57.42	76.56	95.70	114.84	15.81	15.81	15.81	15.81	15.81	15.81
Operations Cost	6.67	13.34	20.01	26.68	33.35	40.02	5.35	10.49*	15.63*	20.77*	25.91*	31.05*
Launch Cost	18.00	36.00	54.00	72.00	90.00	108.00	3.00	6.00	9.00	12.00	15.00	18.00
Module Refurbishment	-	-	-	-	-	-	-	4.91	9.82	15.47	20.38	25.29
TOTALS	133.22	177.03	220.84	264.65	308.46	352.27	108.19	121.24	134.29	148.08	161.13	174.18
						Savings →	25.03	55.79	86.55	116.57	147.33	178.09

* $0.18 launch operations and $0.03 transport saved on not handling total payload; $5.14 million recurring ops.

SAVINGS (CUMULATIVE) WITH REFURBISHMENT ON LOW-COST OAO (6-YEAR PROGRAM)

(Including Benefits of Module/Component Ground Refurbishment)

Fig. 8-30

LMSC-A990556

Cost Elements	Expendable-Launched ($ Million)				Shuttle-Launched ($ Million)			
	Length of Program				Length of Program			
	4 Yrs.	6 Yrs.	8 Yrs.	10 Yrs.	4 Yrs.	6 Yrs.	8 Yrs.	10 Yr.
Non-Recurring Cost	97.99	97.99	97.99	97.99	85.70	85.70	85.70	85.70
Unit Cost of Payloads (incl. 1 backup unit)	100.52	145.20	189.88	234.56	49.15	49.15	49.15	49.15
Launch Cost	127.20	190.80	254.40	318.00	40.80	61.20	81.60	102.00
Operations Cost	27.12	40.68	54.24	67.80	23.90	35.85	47.80	59.75
Module Replacement (Cumulative) (inc. savings by refurb of modules & components)	-	-	-	-	14.16	30.44	44.60	60.88
Totals	352.83	474.67	596.51	718.35	213.71	262.34	308.85	357.48
				Savings	139.12	212.33	287.66	360.87

SAVINGS WITH PAYLOAD AND MODULE/COMPONENT REFURBISHMENT ON 2-YR. SEO

Fig. 8-31

LMSC-A990556

OAO - at 1-Year Cycle

$$\frac{\text{Refurb OAO}}{\text{New OAo}} = \frac{\$5.06M}{\$15.81M} = \underline{\underline{.325}}$$

SEO - at 2-year Cycle

$$\frac{\text{Refurb SEO}}{\text{New SEO}} = \frac{\$3.81M}{\$9.83\ M} = \underline{\underline{.390}}$$

8.5 EFFECTS OF MULTIPLE PAYLOADS

A large portion of the payloads in the NASA mission model do not require the full performance capability or available cargo volume provided by the shuttle or the LCE, while many missions require a Tug or propulsive stage to achieve the required operational orbit. Thus, many flights will carry more than one payload, either multiple payloads of the same program, payload/Tug combinations, or a group of payloads from different projects and possibly to differing destinations. The nature of on-board checkout, the capability of Shuttle crew to perform payload-related activities within certain time constraints, and the safety requirements imposed upon active payload and propulsion stage components must all be considered in developing operational requirements for multiple payloads. Only a cursory study was done of the multiple-payload implications.

8.5.1 Sharing of Launch Vehicle Cost and Volume

It has been fairly common for expendable launches to carry secondary payloads when excess performance was available. Such payloads usually were given a "free" ride on a non-interference basis, and the payload project provided their own adapters, checkout gear and launch support equipment. Communications satellites are good examples of multiple "common" payloads, with as many as eight carried on a single launch, and with a customized "dispenser" integrated with the upper stage. The Shuttle's large cargo bay provides volume for more than one payload, and the single payload may be the exception rather than the

rule. The low-cost OAO, for example, with a payload checkout set installed in the shuttle, occupies only half the cargo bay length and uses less than half the load capacity (ABES in) or less than a quarter the capacity (ABES out) to the 405 nm (750 km) 35° orbit. This excess capacity may be used to carry other payloads, to use a less-than optimum (perhaps standardized) trajectory, to make plane changes, or to retrieve and return a defunct payload after completing the primary mission. Alternatively, there might be a non-deployed payload installed and operated in the sortie mode for up to seven days, at no additional launch cost. Utilizing this potential excess capacity is one way to achieve additional benefit from shuttle launches, but obviously involves long-range planning and funding and considerable flexibility in operations; therefore, making the prime payloads low cost is the principal recommended method for achieving firm cost savings.

8.5.2 Multiple Payloads for Syneq Missions

Payloads destined for synchronous orbits, such as the SEO, come closer to utilizing full shuttle projected capacity, but even these types of missions may be expected to have appreciable excess to place ancillary payloads into LEO or even take several hundred pounds to synchronous altitudes. Section 3.4.2 provides data on some of the tug options; if the largest tug were used with the 3127 lb (1418 kg) SEO, an additional payload of 483 lb (193 kg) could be carried on the round trip mission.

8.5.3 Shuttle Operational Interfaces with Multiple Payloads

The mechanical installation of a payload and a Tug in a shuttle is illustrated in Fig. 2-46, paragraph 2.5.2. Not shown and more difficult to visualize is the servicing and checkout provisions for the tug. The power, instrumentation, command and data interfaces for the tug could be handled exactly as described for the SEO, with a dedicated payload checkout set for each; with good coordination of design, and with the use of additional plug-in units, one checkout set can probably accommodate both. Pre-launch servicing, on the launch stand,

is quite simple for the SEO, but the tug must be chilled down and perhaps 60,000 pounds (27,000 kg) of LH_2/LO_2 must be loaded within a few minutes in an operation closely dovetailed with the loading of perhaps fifty times that much into the shuttle. Propellant fill, dump and vent provisions must be made for on pad servicing or abort and dump provisions are required of the payload by shuttle in the event of an ascent or on-orbit abort. The detailed methods for satisfying this requirement, and the relationship of the deployment mechanism to the servicing connections is a topic requiring more investigation than was within the scope of this study.

8.5.4 Orbit Deployment Considerations

If a number of similar discrete payloads are to be checked out and deployed in the same general orbit plane, they could obviously be: (1) mounted on a dispenser rack which, in turn, would be installed upon the deployment mechanism supplied with the shuttle, or (2) mounted on separate deployment mechanisms. If the payloads are from different contractors or agencies, the former approach would require considerable integration and the building of a special dispenser. A basic alternate approach might be to use suited crewmen, operating IVA, to assist in deployment, using a single deployment mechanism and taking advantage of multiple payload hold-down provisions for securing the payloads during ascent or return. While offering potentially more variety, this multiple-payload cargo mix is expected to prove easier to handle than the payload/propulsion stage combination.

Section 9
LOW-COST PAYLOAD INTERFACES WITH GROUND AND FLIGHT OPERATIONS

Ground or Launch Operations and the ground Flight Operations are recurring-cost operations that can be very substantial for long-duration flights or for low launch rates spread over a long time span. The design of the spacecraft and experiment systems has a significant effect upon the support costs; with many trades possible between on-board complexity vs ground decision-making and control, and between RDT&E and Operations costs. Because of project phasing and interrelationships, shared usage of launch, tracking and data acquisition facilities, mission windows and priorities, orbit altitudes and inclinations, etc., the system decisions are compromises between independent payload system optimizations and integration of the payload system into the total space program. Payload interfaces, then, depend upon many factors -- physical, functional, political, geographical, temporal and procedural. The approach for this study, to put reasonable bounds upon the scope, was to examine three types of payloads, each a derivative of a historical program, with appropriate modifications representative of 1970 technology, and the NASA 1978-1990 Mission Model. The launch site was assumed to be at KSC for either the space shuttle transportation system or the expendable launch vehicles. Communications, tracking and data acquisition, by design, are within the capabilities of NASCOM STADAN or MSFN and are provided by the NASA Office of Tracking and Data Acquisition at no direct charge to the user. (It is recognized, of course, that design of a payload system that imposes heavy loads on the networks, such as continuous coverage, very high data rates and volumes, special ground station sites or requirements for an Orbital Data Relay System, could result in the need for increased OTDA budgetary outlays, and the temptation to optimize payload program costs at the expense of other system elements must be thoroughly evaluated.)

This section describes the interfaces between the low-cost payloads (described in detail in Section 5) and the ground and flight networks. The payload check out set referred to is described in Section 8 of the report.

9.1 LAUNCH OPERATIONS

Ground operations, for the purposes of this study, commence with the shipment to the launch base of a payload or payloads that have been accepted by the government as complete, flight-qualified systems. Each payload is not necessarily complete at this time, but the integration has been accomplished and proper functioning demonstrated; quite often flight batteries, pyrotechnics and kick motors, solar arrays and other bulky, odd-shaped or otherwise difficult-to-handle appendages may be shipped separately or drop-shipped from vendors or subcontractors direct to the launch base. Logistics supplies and spares, including spare payload modules, all ready for use, may be concurrently or separately shipped, so that they are available to support the launch operations. At the launch base each shipment is received, inspected for condition and identification, stored or moved to an assembly and checkout area. Installation of solar arrays, flight batteries, pyrotechnics, etc., and verification of flight readiness is completed prior to mating operations.

9.1.1 Pre-Launch Operations with the Space Shuttle

The use of Shuttle as the launch vehicle considerably simplifies payload launch operations because of the standardized interfaces and the assumption that the Shuttle is a "common carrier" that delivers cargo into the desired orbit. There is no payload fairing that must be separated during ascent and the payload health status can be determined before deployment under direct observation and control of the Shuttle crew. This leads to a reduction in the required assembly, mating and testing activities at the launch base, and the consequent shortening of the time the payload spends at the launch base before launch.

For a Space Shuttle launch, the payload (together with a flight worthiness certificate) is transferred to the Shuttle maintenance/assembly facility about fifteen work shifts (i.e., five to fifteen days) before launch, for pre-mate readiness checks and loading into the orbiter cargo bay. Payload/orbiter combined systems tests, including EMI compatibility, and cargo bay closeout should be completed before start of shift "-10", as shown in Fig. 9-1. This schedule

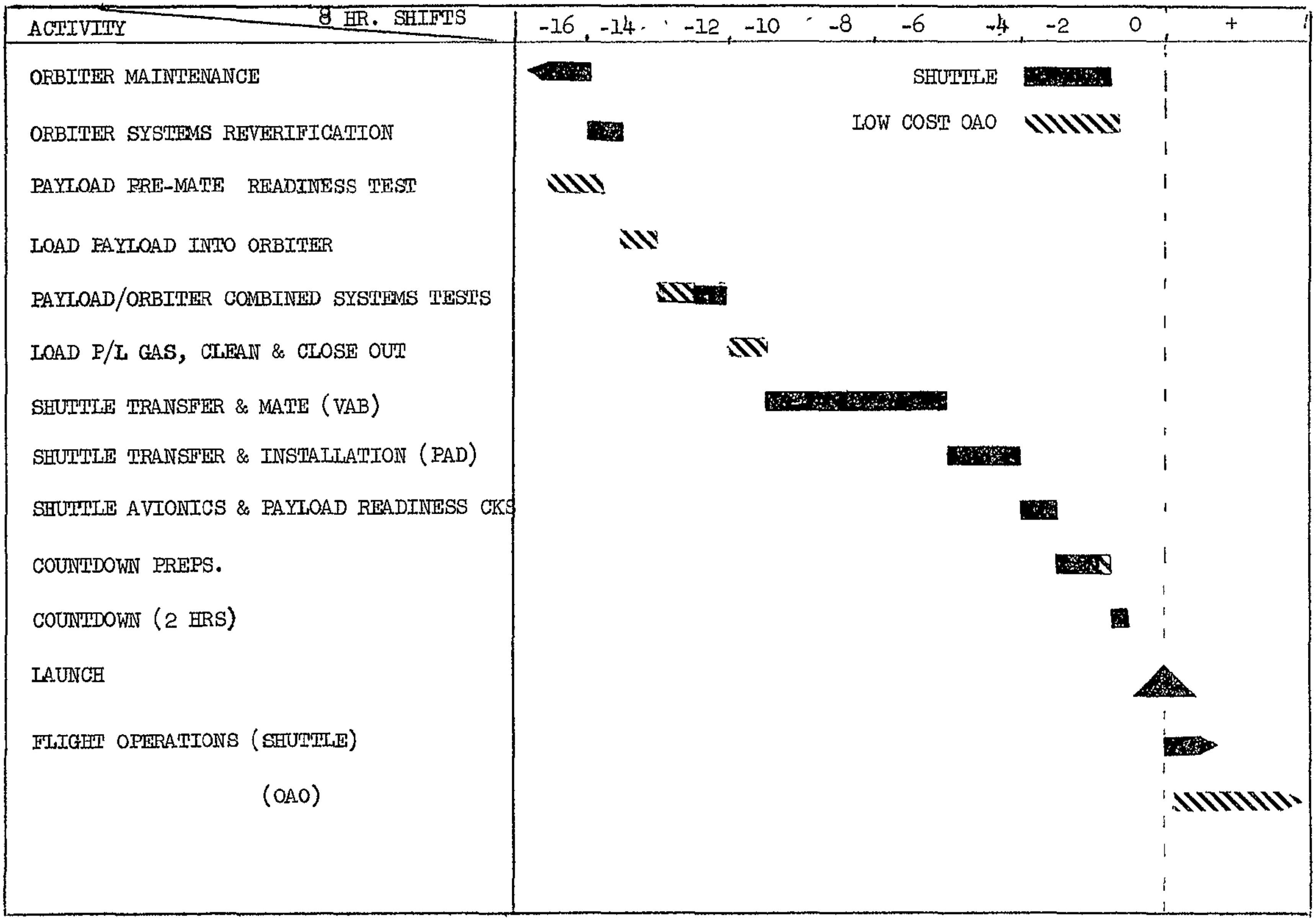


Fig. 9-1 Typical Payload/Orbiter Combined Launch Operations

has been extrapolated from the NASA/KSC LC 39 Space Shuttle Test Plan, TR-1078, 4 Nov 1970. Sixteen hours have been added to the KSC schedule to permit combined systems tests, primarily to assure EMI/RFI compatibility, acceptability and data bus interface, and to allow for loading attitude control gas, cleaning and closeout of the cargo compartment. On the launch pad, provisions are made for a two-hour period just prior to the Shuttle countdown preparation, during which the payload may be checked out through the Shuttle data bus interface. The payload is inactive or in standby condition, with controlled environment, but unavailable to the payload project, until late in the "-3" shift, when a portion of the Shuttle avionics checks may be devoted to interrogating the payload or verifying payload continued readiness. Any payload activities required during the countdown preparations and final (2 hour) countdown will be accomplished by the Shuttle on-board systems and crew. The passive safety monitoring of flight-safety critical payload parameters and cargo bay conditions is accomplished by the Shuttle orbiter on-board checkout and data management system, with the summary results available as required to the Shuttle control center for information.

9.1.2 Pre-Launch Operations with Expendable Launch Vehicles

Combined operations for the case of a low-cost expendable launch vehicle differ from those described above primarily in location and timing of the activities and in greater direct launch support requirements from the payload program. The booster-adapter and payload fairing (assumed to be standard for the particular launch vehicle, hence GFE) are normally installed on the payload in the payload processing facility. The exception to this will occur with the new long Centaur fairing where the fairing covers both the payload and the Centaur upper stage. In this case, the fairing will be installed after the payload has been mated to the Centaur.

The combined package is transported direct to the launch pad for mating to the launch vehicle. The timing is about the same, approximately two weeks (14 or 15 work shifts) before launch. However, the mating, interface compatibility

verification, EMI, and Joint Flight Acceptance Composite Tests are done at the pad, interspersed with all the checkout and preparations for the launch vehicle. Limited open-loop RF radiation is permitted, so primary control and checkout of the payload is accomplished through a payload umbilical system and payload ground support and checkout equipment located on the service structure, in vans, or at a remote control center. In this mode Payload Operations are more extensive, extending up to liftoff, and involve a relatively large crew of engineering and technician specialists capable of rapid detailed analysis and corrective action to avoid launch delays. Access to the payload is limited once the payload fairing has been installed and module replacement requires at least partial fairing removal, and in some cases could require de-mating of the payload from the launch vehicle. Following customary practice, the payload program would supply the air conditioning, power and pressurization equipment to maintain the payload at least until terminal countdown commences.

The net effect, then, of an expendable launch is that more support equipment and a larger launch operations crew is required for the payload than with the Shuttle. The payload activity span is also longer, but this has negligible additional effect upon cost since the payload crew would be kept available until liftoff in either case.

9.1.3 Low-Cost OAO-B Launch Operations

9.1.3.1 Shuttle Launched OAO. For a shuttle launch it is assumed the shuttle operations are independent of the payload program except for the initial payload mating and pre-launch compatibility testing. The OAO-B payload would be shipped by air (Guppy-type aircraft) in a launch-ready condition, fully assembled, checked-out, and accepted, at least one month before scheduled launch. (For the first flight of the QTV two months will be required for pre-launch familiarization and activities.) A standard payload test set accompanies the payload to the payload servicing area, either part of the Shuttle facility or a separate facility. Here, the payload is inspected for shipping damage, tested, alignment checked, and cleaned. Shuttle mounting provisions and interfaces, procedures and schedules are verified. At the scheduled time the payload

is moved to the Shuttle loading area, interfaces are verified, and the payload is installed into the Shuttle cargo bay. Payload/Shuttle combined system tests are conducted to verify compatibility, with particular emphasis upon EMI/RFI. The payload now becomes quiescent while Shuttle launch preparations continue. After Shuttle is rolled out to the launch pad and installed thereon, perhaps 3 days to two weeks from payload loading, a launch servicing and countdown cycle commences. Early in this cycle a limited time, probably less than two hours, will be allocated to a payload flight readiness verification. This corresponds to the OAO "aliveness" test. The payload launch crew, except for a small launch base cadre, is made up of project, subsystem and test personnel who have participated in the assembly and system testing and have followed the "launch-ready" payload to the launch base. Payload health status is monitored through the Shuttle data bus and communications system except during Shuttle propellant loading, beginning at T-2 hours, during which all systems not involved in propellant loading are secured. Upon Shuttle liftoff the launch operations phase for the payload is considered completed.

The launch operations crew for the low-cost OAO is sized to provide on-the-spot capability for analysis, evaluation and correction of anomalies or contingency operations, in addition to the routine verification, test and interface functions. This is considered conservative, since real-time communications with the test and integration facility and the OAO control center make available a large reservoir of specialists and access to the project data banks.

The payload launch base personnel cadre, supplemented if necessary by specialists from the project office, are expected to be able to handle logistics missions involving Shuttle maintenance missions for replacement of complete modules in the low cost OAO or recovery and return from orbit. Returned modules or payloads would be shipped to the field depot facility for refurbishment, analysis or disposition.

9.1.3.2 Expendable-Launched OAO. If the low-cost OAO-B is to be launched from an expendable vehicle, the TIII-L2 is to be used; in this case there would be

no preliminary QTV flight, and more extensive ground qualification testing would be required. Activation of payload operations at the launch site, except for liaison planning and technical integration, may then be delayed until about two months before the scheduled launch. On-pad buildup of the launch vehicle (or at least of the mating of the payload to it) is assumed, and the general discussion in 9.1.2 applies. Because this is a one-flight program (baseline) without opportunity for orbit pre-deployment checkout or return of the payload from orbit for diagnosis, testing will be more rigorous, with tighter limits, repeated more often, and results of each test compared to previous tests for trend analysis. The use of a standard payload test set is highly desirable as it helps insure consistency in test methods, procedures and equipment, and is essential for valid fault analysis trend data and high confidence. All payload activities subsequent to the start of mating operations must be closely dovetailed to the countdown schedule, which will allocate time periods for payload activities, for RF silence or radiation, for joint testing, and for complete exclusion of payload personnel from the launch pad. Activities during the pad operations phase are usually scheduled around the clock, so that the payload crew must be large enough to provide this kind of coverage during the week to ten days the combined vehicle is on the pad. In this respect, launch operations for the low-cost OAO-B do not differ markedly from those for the baseline OAO-B, launched on an Atlas-Centaur; the primary gain is simpler checkout because of the greater autonomy provided by the on-board computer in the low-cost OAO payload.

9.1.4 Synchronous Equatorial Orbiter (SEO) Launch Operations

9.1.4.1 Shuttle-Launched SEO. The SEO mission plan requires each satellite to be delivered to a synchronous equatorial orbit station at a preselected longitude in order to provide the required earth coverage for the mission. The combination of a Space Shuttle and reusable Space Tug satisfies the placement requirements in an economical manner provided the Tug has sufficient performance for round-trip placement and retrieval; this means replacing and retrieving a malfunctioning SEO for refurbishment as well as the initial SEO

placement on station. Shuttle and Tug operations are independent of the SEO program operations except during the brief pre-launch joint flight acceptance composite tests (J-FACTS) and the coordination during on-orbit tests with the Shuttle orbiter prior to separation. The use of a Standard Payload Test Set installed in the Orbiter cargo bay to check out both the Tug and the SEO requires that both be designed with this capability as a firm requirement; the versatility and cost advantages are such that this requirement should be considered valid.

Five SEO spacecraft are provided for a 4-station system. Following the Lunar Orbiter precedent, a backup launch-ready spacecraft is required for each initial-placement launch. Should schedules prove too tight, this requirement could be waived, thus gaining two months for the manufacturing, integration and test activities. Because of the repetitive launches and the need for maintaining readiness for replacements, the launch operations activity for SEO is planned to continue for the program duration.

The SEO satellite is air-shipped from the manufacturing/integration facility to KSC in a nearly launch-ready condition approximately six weeks before scheduled launch. For convenience in handling, the solar arrays are shipped separately. Upon arrival each SEO is inspected for possible shipping or handling damage. After completion of solar array and battery installation and expendables loading and after complete integrated system tests, the camera module is removed, loaded with fresh film and bimat, and reinstalled. The SEO is then moved to the Shuttle Payload Facility and mated to the Tug. Interface compatibility and EMI/RFI tests verify the joint operation of this combination and readiness for loading into the Shuttle Orbiter. Approximately 15 days (based on one shift per day operation) before launch the payload/Tug is moved to the Shuttle loading area. Interfaces are reverified, the payload/Tug is loaded into the horizontal Orbiter, and complete interface, EMI/RFI and joint flight acceptance composite tests are performed. The Orbiter is then moved to the VAB, erected and mated to its Booster on the launcher. The complete vehicle is then transported to the launch pad, umbilicals are installed and final avionics operational tests are performed, and countdown preparations are completed. During

the final few hours of countdown the Shuttle and Tug are loaded with propellants, while the payload is in a standby mode with continuous safety monitoring by the Shuttle on-board checkout systems.

The GSE required to support launch operations should be copies of equipment used for factory handling, integration and test. The use of a GFE standard Payload Test Set (or a program-supplied equivalent) is assumed. In addition, the following is a typical GSE List:

2 Vehicle Handling Dolly
1 Solar Array Handling & Installation Fixture
2 Vehicle Transporter (may cycle to factory for new SEO)
1 Photo Module Handling & Installation Fixture
1 Set of Work Stands
1 Photo Module Transportation & Storage Unit (4 in pipeline)
4 Equipment Module Transportation & Storage Unit (6 in pipeline)
4 Solar Array Transportation & Storage Unit (4 in pipeline)
1 Micropositioning Hoist Unit
1 Gas Pressurization Unit
1 Display, Control and Timing System
1 Wideband Video Tape Recorder
1 Set of Test Cables
1 Set of Slings & Spreaders

The combined launch operations for SEO differ from those described for OAO in that the Tug is an extra element in the system. Figure 9-2, prepared on the same basis as Fig. 9-1, shows a launch timeline for a shuttle-launched SEO.

9.1.4.2 Expendable-Launched SEO. The low-cost SEO may also be deployed by an expendable launch vehicle such as the Titan IIID/Improved Centaur providing the Centaur is capable of a triple-burn, long coast mission necessary to place the SEO into circular equatorial orbit at the assigned longitude for each satellite. Since the four satellites are assumed equally spaced, this means long

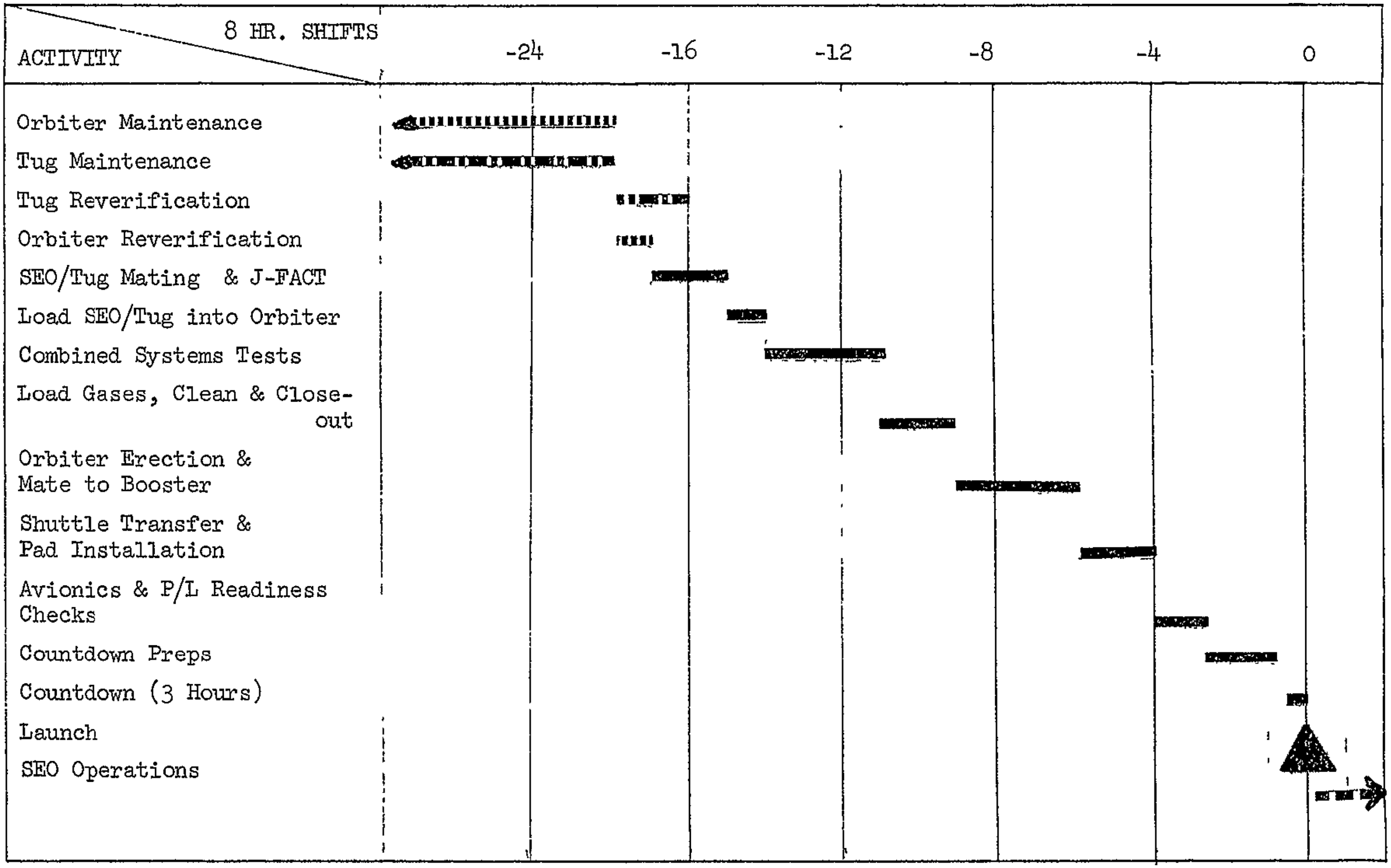


Fig. 9-2 Launch Timeline - Shuttle-Launched Low-Cost SEO

waits in LEO or several revolutions in a 10.5 hour transfer orbit to achieve some stations. The low-cost SEO does not have propulsion capability for injection or station change. The drift mode for achieving the desired location is not provided in the assumed mission plan; the Shuttle/Tug provides precise station placement. As with Shuttle, the payload launch operations are not so much affected by the particular vehicle configuration or upper stage selection as they are by the buildup and launch timelines. If the use of a Centaur Standard Payload Fairing is assumed, the discussion on expendable launched OAO apply to the SEO as well. It should be noted that one interface is eliminated, namely the SEO-Shuttle, while the SEO-Tug interface is equivalent to the SEO-Centaur interface for the expendable system.

9.1.5 Small Research Satellite (SRS) Launch Operations

The SRS is intended as a secondary payload, carried on a Shuttle flight for which the primary mission does not require all the Shuttle capability. The HIGLO mission (described in Section 3) was used as the SRS baseline; this requires continuous observation in a polar, eccentric sun-synchronous orbit for a period of one year or more.

A Shuttle launch would carry three satellites into a nominal 100 nm (185 km) parking orbit, checkout two for ascent survival, using the third (a spare) as a replacement if necessary; Shuttle would then restart and inject into a 100 x 150 nm (185 km x 278 km) transfer orbit, with proper timing to provide a noon "line-of-nodes" when the SRSs are dispensed at apogee. The SRS is spun up by integral spin rockets and its solid rocket motor is fired to place it in a 150 x 275 nm (278 x 509 km) orbit. The second SRS is also placed in orbit, after checkout, in a passive storage mode until required to replace the first one. Launch operations are similar to those described for the Shuttle-launched low-cost OAO, though much simplified and dovetailed with the primary mission cargo loading operations; no direct participation by the SRS program personnel is assumed after delivery of the flight-ready SRS to the Shuttle loading area.

The low-cost SRS may also be launched by an expendable such as the 3 Segment SRM/Titan Core II/Agena (LCE) or the SLV-3C/Burner II. Two SRS would be carried, although the LCE has the capability for four. As with the Shuttle, launch is assumed GFE. The launch vehicle adapter rack is supplied with the payloads, and mating occurs at the launch pad, very similar to the expendable-launched OAO.

9.2 FLIGHT OPERATIONS

Liftoff normally marks the commencement of flight operations for an expendable-launched payload, but with a Shuttle launch they may be deferred until the Shuttle is in orbit and is preparing for payload deployment. Further, if the Shuttle carries an onboard payload test set, the ground-based flight operation may take over control of an "operational" satellite instead of having to provide peak resources to initialize and thoroughly checkout all modes of operation. Assuming the same degree of autonomy and the same "station" locations, this launch and initial checkout represent the prime differences between flight operations for the two types of launch systems; a recovery mode, applicable to some Shuttle-launched payloads, does, of course, add to the rendezvous, docking and safing operations which, optionally, may be under either ground- or shuttle-controlled.

9.2.1 Low-Cost OAO-B Flight Operations

Preparations for flight operations of the low-cost OAO payload are scheduled to commence approximately 16 months before the scheduled first launch (the QTV flight) with planning and scheduling activities, establishment of requirements and coordination with similar functions of the Office of Tracking and Data Acquisition (OT&DA). The basic plan is patterned after OAO-A2 operation plans, modified by the technology growth and changes in the low-cost OAO flight system. Use of the OAO control center, SPCS, NASCOM and STADAN is postulated; the same RF channels have been selected, but command and telemetry formats differ. The incorporation of an on-board computer in the new low-cost OAO

design materially decreases the amount of SCPS detail computation and preplanning that must go into formulation of command messages. The simpler and fewer command messages and the ability of the payload to store up to two days' worth of commands decreases the up-traffic through NASCOM and STADAN so that continued use of all the five stations currently supporting OAO is not required. This permits scheduling the use of these stations to support more other programs than currently possible. (Alternatively, it might permit economies by reducing the scheduled support from some of the remote stations to a one-shift basis if OAO were the sole user, a situation not presently true.) Only software changes are required for the remote station support of low-cost OAO, as the basic design restrictions provide for a system compatible with the Shuttle avionics, standardized payload test set and STADAN. Because tracking and data acquisition during the Shuttle ascent phase is not a responsibility of the payload project and because the payload may be checked out on board the Shuttle before deployment, and checked out with the ground stations before separation, much of the risk and high peak load upon flight operations during the first two weeks of flight is eliminated. Assuming that the control center will continue to be required to operate 24 hours a day, a basic staff of 28 is planned; this includes NASCOM and STADAN coordinators on each shift. Operations computer programming and Support Computer Programming System support is estimated to require a 26 man-level of efforts and Experiment support a 10 man-level. Flight operations planning, procedure changes and status tracking will taper down from a 5 man peak to a 2 man sustaining level. STADAN support is estimated to require an increase in the T&DA manning level of 40 men over and above their requirements if they were not supporting the OAO program. (This is normally covered in the T&DA budget and hence is not considered a payload program cost.)

Operations for the expendable-launched low-cost OAO accrue the same advantages from greater autonomy as the Shuttle-launched system except for a heavy peak of deployment, checkout and initialization diagnosis and operation extending from launch for perhaps two weeks, and involving some supplemental (but existing) ground stations.

9.2.2 SEO Flight Operations

A Shuttle launched SEO is initially launched due East and the Orbiter injects into a parking orbit approximately 6 minutes later. (Alternatively, the Shuttle could inject into a 45 or 50 nm x 100 nm (83 or 93 x 185 km) transfer orbit, and go into a 100 nm (185 km) parking orbit approximately 51 min after launch.) After the orbiter health and operating conditions have been verified the payload is checked out, using a Standard Payload Test Set operated by an orbiter payload crew. The Tug will normally be checked first, and if it has survived the launch and ascent satisfactorily the SEO would then be checked out. The Tug/SEO is then deployed on booms and ground command/communication with each element verified. After verification of test and communications data the Orbiter separates to a safe distance away from the Tug/SEO. The tug reorients to ignition attitude and, on time in proper phase, injects the combination into the transfer orbit. Injection will be timed to assume that the transfer orbit apogee corresponds approximately with the desired SEO placement longitude. At apogee, the tug will perform the required plane change and orbit circularization maneuver. Flight operations could start as early as the initial payload checkout in the orbiter cargo bay, or could be deferred until achievement of final placement at the correct longitude in synchronous equatorial orbit. After verification of command acquisition by the primary tracking station for that location, the SEO will separate from the tug and continue operations under ground control, while the tug stays in the vicinity for up to 24 hours during which SEO performance is evaluated by quick-look data at the tracking station and data relayed by ground and/or COMSAT links to the control center (at NASA/Goddard). The Tug then returns to LEO for rendezvous with the waiting Orbiter and return to earth. (The Tug has the capability of bringing SEO back with it if operations are not satisfactory.)

A dedicated control room equipped to handle five SEO satellites is part of the Mission Control Center, assumed to be located at the GSFC, Greenbelt, Maryland. Four existing NASA ground stations will each provide an S-band two-way link to its assigned SEO; a fifth, the KSC station, is used for pre-launch and launch

checkout and as a backup to the Rosman, North Carolina, station for LEO checkout. Station operations and communications from the remote sites to MCC is provided GFE by the NASA Office of Tracking & Data Acquisition. Some SEO-peculiar equipment and a small cadre of SEO specialists will be located at each station, thus providing the capability of continuing SEO operations during temporary interruptions in the communications links to the MCC.

The initial operations plan placed the four satellites at 095°W, 05°W, 085°E, and 175°E, using STADAN stations at Rosman, N. C., Madrid, Spain, Carnarvon, Australia, and Kauai, Hawaii, respectively, with 30-ft (9 km) diameter or larger ground antennas. Since this placement was arbitrary, locations could be chosen that would allow one ground station having multi-link capacity to handle two SEOs, thus reducing the total number of ground stations to three. For example, satellites at 37°W and 128°W could utilize Rosman, with the other two at 53°E and 143°E, serviced by Madrid and Canberra, respectively. The ability of Shuttle to hold in LEO for desired phasing makes placement at any desired longitude feasible. Further flexibility in flight operations may be achieved when Tracking & Data Relay Satellites are available, but basic SEO functions are not materially affected.

Flight operations for an expendable-launched SEO commence with launch, primarily monitoring payload status, either by direct communication or by status reports from the launch vehicle flight operations. Flight plans are much tighter, with coast and stay times limited by the upper stage characteristics. This makes some longitudes easily obtained and others require complex operations by the launch vehicle. Once on station the operation of the SEO is the same as previously discussed, except that the satellite is stationary and hence always in range of the same ground station. It is not even necessary that a continuous communication link be maintained with the SEO (for the mission defined), making the concept of servicing more than one satellite from one ground station all the more attractive. Similarly, a single control center can easily handle the routine spacecraft health monitoring and control and the experiment data flow from the whole project satellite network; a separate control room or

crew for each satellite is unnecessary. Further, through the use of communication satellites, already employed by NASCOM for some locations, even video data can be relayed to the GSFC control center in near-real time when necessary, further reducing the need for special crews and equipment at remote locations.

9.2.3 SRS Flight Operations

The shuttle-launched SRSs (one to be active and one to be stored on orbit for a nominal six months and then to take over the active function) are checked out in the cargo bay with a payload test set, and the ground station and control center interfaces verified by direct RF open loop checks. The Shuttle then injects into a 100 x 150 nm (185 x 278 km) transfer orbit and deploys the two SRS. Control of spin-up and SRS injection into the final 150 x 275 nm (278 x 509 km) sun synchronous near-polar orbit may be controlled from the Shuttle or from a ground station if the injection point is within acquisition range of one. The SRS has been designed to use VHF telemetry and command, so all of the existing STADAN facilities may be considered available in developing the mission flight plan. The SRS is nearly autonomous as far as housekeeping functions are concerned, so that some nominal schedule like once a week (or about every 100 orbits) when the spacecraft is over Rosman, for example, is sufficient for ground update, such as spin axis correction or stored program changes. Experiments are not the kind that require continuous coverage, and can, in general, be programmed from the ground so that they are active in the desired revolutions or portions, and the data readout when within range of a ground station; 120 min of data recording are provided, or about $1\frac{1}{4}$ orbit revolutions, and this can be readout in 7.5 min at a 6.3 kbps rate, processed by the ground station and relayed at lower rates to the GSFC control center servicing the SRS program. There it may be processed or passed on in raw form to the experimenters. The flight operations are low key, so that one control room may service a number of similar programs. The use of Tracking and Data Relay Satellites would remove all constraints on continuous experiment operation imposed by ground station locations, hence enhancing the availability and flexibility of the SRS for other types of experiments.

Flight operations for the expendable-launched SRS differ from the Shuttle-launched mode in two respects: (1) there is no spare "third" spacecraft that can be substituted prior to deployment; (2) on-orbit checkout prior to deployment must also be controlled from the ground so the two satellites should be monitored from liftoff, on shared or separate channels, or through the Agena telemetry. The Agena upper stage of the new low-cost expendable has the capability for following a deployment sequence similar to that described for Shuttle, including phasing and positioning the SRS at deployment. These differences have only minor effect on the cost of flight operations.

9.3 IMPACT OF LOW-COST PAYLOADS UPON FUTURE LAUNCH AND FLIGHT OPERATIONS

Ground operations in support of pre-launch and launch activities, ascent, orbital maneuvers, placement, and experiment or mission operations will be impacted both by the choice of expendable or shuttle launch vehicles and by the implementation of total system-optimized low-cost methods and techniques for design, qualification, manufacture, checkout and operation of payloads. Today's launch bases, control centers, ground station and communications networks, and computer facilities need not be constrained to preserve the status quo, but may plan and implement efficiencies and improvements of technical and economic nature on a phased basis with the expectation and assurance that the users-- the payload programs--will be able to continue operations without vital interruptions. The use of shuttle, with it's large cargo capacity, orbital stay time capability and ability to retrieve and return payloads, or the alternative LCE vehicles that also have appreciable performance capability, permit relaxing the conventional weight and volume constraints upon payloads, leading to the low-cost designs described in previous sections of this report. These low-cost payloads, in turn, permit reduction in: (1) launch base handling, assembly, checkout, integration, and GSE development spans and expenditures, and (2) the number of ground stations, communications links, control centers and support facilities needed for maintenance of satellite health, control and data acquisition. These low-cost payloads may be characterized by the operational attributes of autonomy,

adaptability, simplicity, maintainability, and durability.

9.3.1 <u>Autonomy of Operation</u> - is a characteristic of the low-cost payload approach achieved by the use of on-board computers, large memory for stored commands, automatic implementation of redundant or alternate modes upon failure, loss of lock, etc., high data storage capacity and versatility for adaptation to changes in requirements.

The impacts upon launch operations are: (a) <u>the payload has fewer external interfaces</u> to contend with (comprising a mechanical mounting to deployment mechanisms and an electrical interface to the payload test set or vehicle data and power busses); (b) <u>simpler checkout</u> because of self-test capability, modularity, and use of automatic checkout techniques with consistance of checkout equipment and readout data from factory acceptance testing thru to actual operation; (c) <u>less ground support equipment</u> because of the high degree of self-sufficiency and the completeness of the payload test set functional interface; (d) <u>independent test scheduling</u> because the checkout may be accomplished in an RF Closed Loop mode by the PTS while other launch vehicle and range activities are proceeding normally.

The impacts upon flight operations are: (a) <u>continuous monitoring not necessary for</u> health status and spacecraft operations; (b) <u>fewer commands</u> need be generated, verified, dispatched, transmitted, and acknowledged; (c) <u>fewer parameters are monitored</u>; (d) <u>longer intervals between contacts</u> are permissable, making it feasible to control and operate a number of payloads from a single control center with simple scheduling and freedom to reassign ground stations and equipment, including utilization of communication satellite links and TDRS when available. The reduction in number of contacts, amount of routine data handled, number of ground stations needed for initial checkout and routine operation, and the greater link margins permit handling more payloads or missions with the same people and facilities or effecting economies by reduction in the scope of operations.

9.3.2 Adaptability of the Low-Cost Payloads - is the designed-in capability to improve, update, or change subsystems by substitution of equipment modules or by software program modification. The impact upon launch operations is obtained by adapting installation and checkout procedures to the varying cargo mixes that may be expected when Shuttle launches become common. A large variation in scheduling of installation and checkout may be accommodated, and checkout equipment may be shared among other payloads. This same impact carries over into flight operations, making it possible to change mission parameters and schedules without major hardware or launch operations perturbations, simplifying scheduling of network resources, allowing updating of data formats, operating methods and network configurations even in the middle of long-lived programs. In particular, the introduction of TDRS and gradual phasing out of some of the existing ground stations may be accommodated even though important changes may be required in channel frequencies, antenna pointing, addressing and message formatting may be necessary.

9.3.3 Simplicity of Equipment Installation and Functional Isolation - are characteristic of the low-cost payload approach. Support operations, particularly at the launch base, become straight-forward and often may be performed by general technicians instead of requiring large numbers of highly-trained specialists. Repair is usually by module replacement, checkout is automatic, and assembly or installation of such items as batteries, solar arrays, film canisters, etc is uncomplicated by use of ruggedized construction, low-density packaging, and liberal tolerances. Go-no-go end-to-end testing with automatic checkout equipment (payload checkout set) further reduces the complexity of the payload launch operations, permitting manning and scheduling economies. Flight operations are similarly impacted, especially in the areas of operations planning and scheduling and in the evaluation of data and formulating of responses or commands. Fewer specialists are required for routine operations, training is minimized, and the data bank and analysis of equipment performance are less complex than with conventional payloads. Flight operations tasks will probably be reduced to simple routines so that the same personnel may handle a number of different payloads.

9.3.4 Maintainability and Refurbishability - have been somewhat unusual attributes to expect of a payload, traditionally, but are important characteristics of the low-cost payload concept. The proposed payloads are cost effective for launch operations with either LCE or shuttle launches, and offer potentially large bonuses for on-orbit repair, refurbishment and/or modification operations. The amount of logistics support required at the launch base is minimized by the modular approach and the automatic checkout techniques. Faster repair, equipment or experiment replacement, and calibration may be achieved with fewer specialists and less equipment, reducing the risk of delaying launch schedules. The need for project-peculiar handling, test and servicing equipment is minimized, and the problem of personnel training is simplified. The impact on flight operations is less obvious, primarily taking the form of permitting on-orbit repair/refurbishment for some missions, and providing faster replacement for a failed unit.

9.3.5 Durability or Ruggedness - is a characteristic of low-cost payloads made possible by the elimination of tight weight constraints and achieved both by design intent and as a result of increased safety margins or certain equipment. Rigid structures, strong interfaces, integral lifting and handling provisions, modular construction, use of commercial materials, and inclusion in the design of protective covers or seals when necessary all help reduce the complexity and extent of launch operations, simplifying handling and reducing repair and servicing times. Lower skill levels and fewer project-trained people are thus required to support a launch. Flight operations are only slightly impacted by the payload ruggedness; simpler deployment of folded elements, the use of pre-deployed devices and structures, and the ability to employ faster and simpler docking operations provide more operational flexibility but do not materially affect control center or network support requirements.

Section 10
CONCLUSIONS AND RECOMMENDATIONS

The Payload Effects Analysis Study has exposed the source and magnitude of potential payload program cost savings. A credible data base has been established Techniques for low-cost payload design, development, production and operation have been innovated. However, more work is required; and this work should be initiated soon to allow matching of payload development schedules with Shuttle/Tug development. This will assure attainment of the overall savings potential. Implementation of these low-cost payload concepts may prove to be the principal economic basis upon which the Shuttle development can be justified.

The principal conclusions drawn from the study are:

- Significant cost benefits can accrue from new payloads designed to low-cost criteria. The principal Payload Effects* result from:

  (1) Refurbishment of Payloads
  (2) Mass and Volume Constraint Relaxation
  (3) Risk Acceptance

- Some of these savings can be obtained prior to introduction of Space Shuttle and Space Tug if applied to current expendable payloads.

- Additional significant savings in payload development costs are possible with standardization of subsystems and/or use of standard spacecraft.

- With forecasted NASA budget limitations, it is very important to continue the vigorous pursuit of payload cost reductions. These reductions - Payload Effects - can contribute materially to making the Shuttle and Tug economic realities.

* "Payload Effects" = cost savings resulting from low-cost payloads and their usage.

- The savings estimated during this study are reasonably conservative. As experience is gained with implementation of the low-cost approaches, further cost savings are very probable. These savings could be converted into additional space utilization and applications.

This final section of the report will discuss these conclusions. The principal Payload Effects are quantified and the overall impact of standardization upon the NASA funding is described. Of particular interest is the prediction that implementation of standard subsystems could result in a reduction of the NASA projected peak funding for space programs by 16 percent.

The applicability of the study results to other payload programs is discussed and finally, LMSC's recommendations for future activities are presented.

10.1 PAYLOAD PROGRAM COST EFFECTS

This sub-section summarizes and quantifies the primary low-cost payload program effects of weight and volume relaxation; risk acceptance; and retrieval, refurbishment, and reuse.

10.1.1 Primary Payload Effects

During the study, principal cost savings were determined to result from:

- Refurbishment of Payloads
- Weight and Volume Constraint Relaxation
- Risk Acceptance

Although major cost savings were made possible by application of 1970 technology, it is not a true payload effect which is equally available and applicable to all new-start programs. Savings in the low-cost designs with respect to the baseline designs and attributable to technology effects were determined and isolated so that they could be separately considered.

Weight and volume effects were calculated by comparison of total program costs for a fixed program duration (10 years) of the low-cost expendable-launched payloads with the baseline payloads.

Risk acceptance includes Shuttle/Tug-associated increase-confidence (due to in-orbit checkout) benefits of reduced testing, reduced redundancy, reduced sustaining engineering, etc. Risk acceptance effects were determined by comparing total program costs of low-cost expendable-launched payloads with Shuttle-launched payloads (excluding refurbishment).

Applying the refurbishment factors determined during the study, total program costs, including refurbishment, were calculated and equated against the baseline, LCE, and non-refurbished Shuttle modes. The savings attributed to refurbishment are shown graphically in Fig. 10-1.

For a ten-year program span, using the limited sample size offered by OAO-B and SEO, the Payload Effects (payload program savings, excluding any launch vehicle effects) are:

Relaxation of Weight and Volume Constraints	18 ± 3%
Risk Acceptance	9 ± 1%
Refurbishment of Payloads	21 ± 2%

Total payload program savings due to payload effects excluding the new space transportation savings, can be approximated as:

Low-Cost Expendable Programs	18 ± 3%
Space Shuttle/Space Tug Programs	49 ± 4$

It should be noted that the savings resulting from relaxation of weight and volume constraints are available presently if space program planners would consider the use of larger boosters. However, this concept should be examined cautiously on a total program basis as the increased cost of booster procurement and operations could very well exceed the Payload Effects savings benefits.

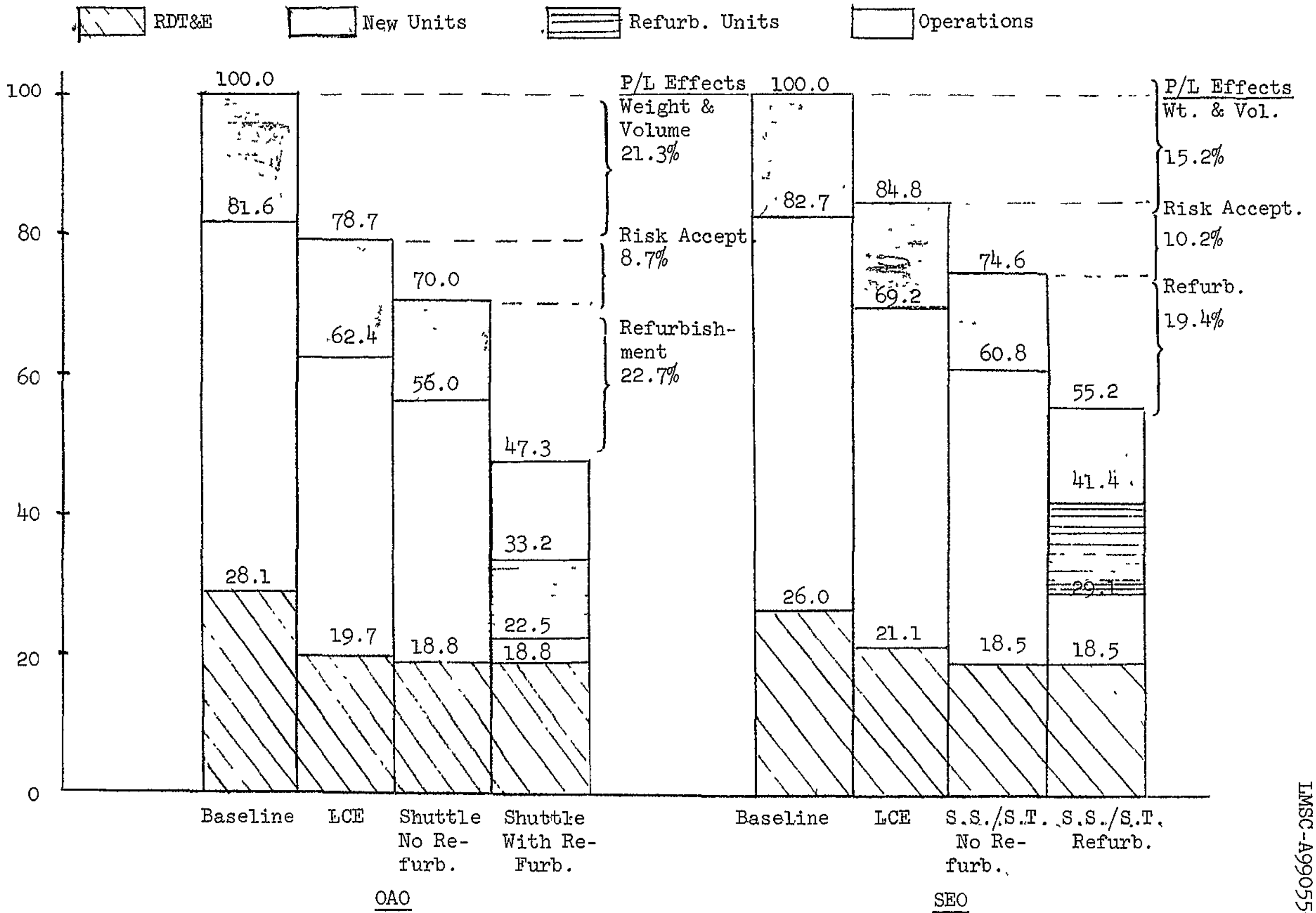


Fig. 10-1 10-Year Payload Program Cost Comparison (Excludes Launch Vehicles)

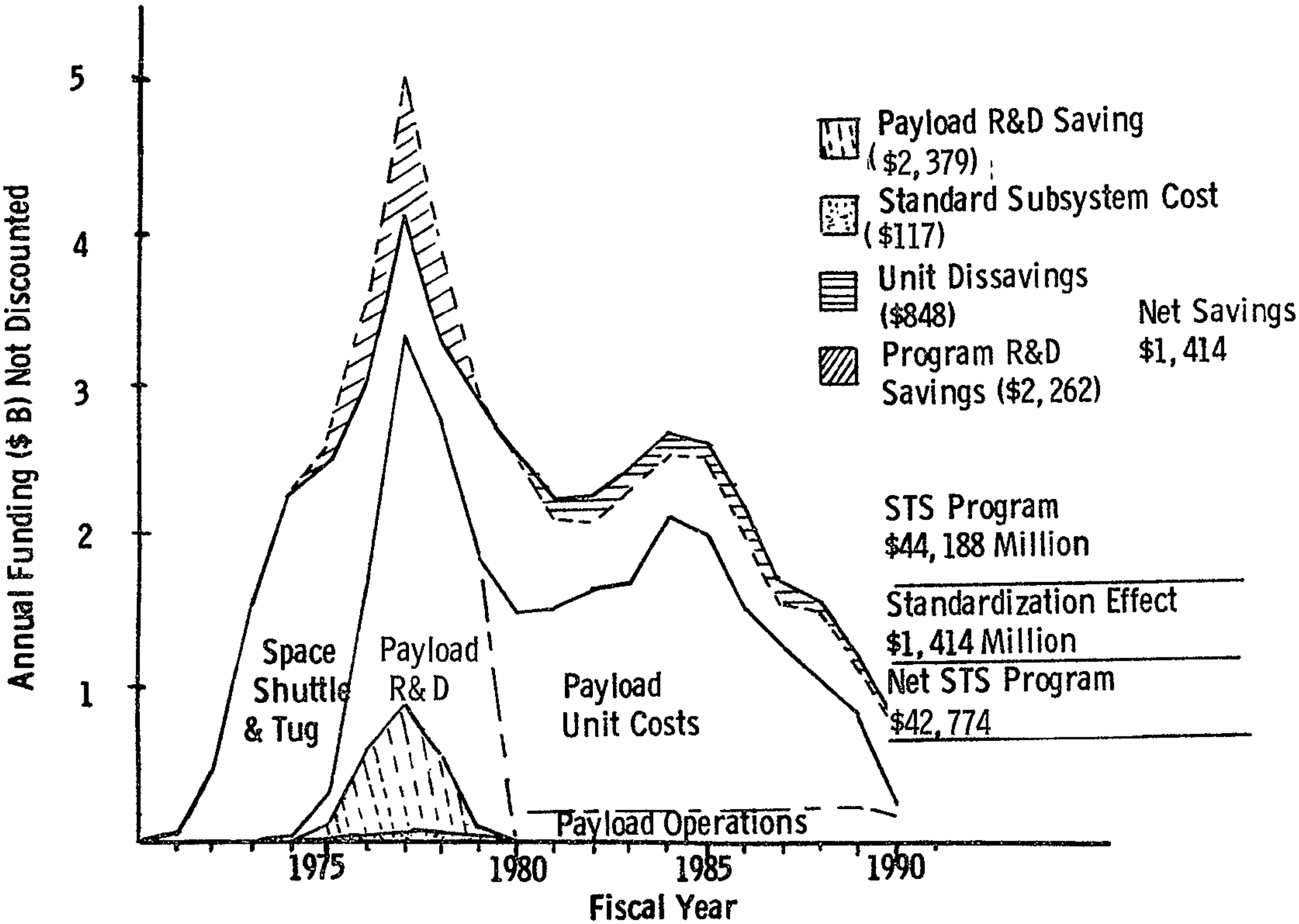
This point was made apparent by inspection of a ten-year SEO program using LCEs compared with the baseline; the added transportation cost resulted in the total "low-cost" program being 2 percent more costly than the baseline.

The importance of retrieval, refurbishment and reuse can be further seen from inspection of the total program costs, including transportation, with and without refurbishment. The savings provided by payload refurbishment approximate or even exceed the cost of Shuttle transportation alone. For example, on OAO-B cost savings for a ten-year program offered by refurbishment are $136 million, while the Shuttle transportation cost is only $46 million. On the ten-year SEO program, refurbishment savings are $90 million while the Shuttle transportation costs are $102 million.

10.1.2 Standardization Impact upon Total Funding Requirements

Results of the study of the standard spacecraft concept showed that development of specific standardized subsystem families with various options of capability can materially reduce overall payload RDT&E costs. Implementation of these concepts can eliminate an estimated $2.4 billion for an investment cost of under $120 million. Using the total space program funding data provided by Aerospace Corporation in Vol. III of the Midterm Report of the Integrated Operations/Payloads/Fleet Analysis Study as a basis for analysis, it was determined that the peak projected funding of $5 billion in 1977 could be reduced to slightly over $4 billion by application of this saving.

This is shown in Fig. 10-2. The solid lines represent the funding requirements without standardization savings. Payload RDT&E was assumed to be completed by 1980 after which time the payload costs are comprised of recurring hardware and operations. Payload operations were assumed to amount to $200 million per year. Development of standard subsystems was assumed to commence in FY 1975. The $2.4 billion savings and the $117 million new RDT&E cost were budgeted over a five year development span using a 50:50 cost spread. It was further assumed that a 10 percent overkill penalty in unit recurring costs would have to be paid. This is shown as the increase (dissavings) in



STANDARDIZATION IMPACT ON TOTAL STS FUNDING

Fig. 10-2

total program funding in the 1980-1990 period. It should be emphasized that this funding picture does not include the impact of refurbishment and reuse upon unit costs, nor does it include an application of social discount. Further, conservatism was provided by estimating that experiments (mission equipment) included in the payload costs amount to 33 percent of the payload costs, and that standardization was applied only to the spacecraft portion of the payload funding.

Inspection of Fig. 10-2 indicates the need for early implementation of the standardized subsystem or spacecraft concepts in order that peak savings may coincide with the peak funding years. On this basis, LMSC strongly recommends that the following schedule of implementation be included in NASA's advanced program plans.

	Start (FY)	Complete (FY)
Phase A - Conceptual Design	1972 - 1st Qtr.	1972 - 4th Qtr.
Phase B - Preliminary Design	1973 - 1st Qtr.	1973 - 3rd Qtr.
Contractor Selection	1973 - 4th Qtr.	1973 - 4th Qtr.
Phase C - Detailed Design	1974 - 1st Qtr.	1974 - 4th Qtr.
Hardware Go-ahead and Start of Phase D - Development and Operations	1975 - 1st Qtr.	-
FIRST FLIGHT APPLICATION	1979	-

Fig. 10-3 Standard Spacecraft Implementation Schedule

10.2 APPLICABILITY OF RESULTS TO OTHER SPACE PROGRAMS

During the study, only a very small sample of the total NASA unmanned space missions was examined in detail. However, it is possible that scaling factors could be derived which would be valid generally for the majority of the future missions where refurbishment is practical. For those missions where refurbishment is not practical, the general savings, resulting from relaxation of weight and volume constraints and risk acceptance still can be applied. While it is known intuitively that similar economies can be achieved with manned space operations, the data resulting from the current study strictly apply only to unmanned payloads.

During the course of the study, some of the more significant trades were implemented; e.g., cost of payload sophistication in terms of weight and volume vs transportation costs, and payload reliability vs orbital maintenance. Furthermore, several approaches to spacecraft standardization on the basis of recurring systems requirements have evolved and have been evaluated.

Resulting sets of economic scaling factors have already found application in the estimating of potential program-wide cost savings for unmanned missions. The study results can be generalized in the sense that they are typical of any industrial endeavor gaining maturity. As interfaces and interactions become clearer and as the cost drivers become identified, positive action can be taken in the overall optimization process. However, more specific extension of study results to other unmanned and manned missions will require further study.

10.3 RECOMMENDATIONS FOR FUTURE PAYLOAD STUDIES

The Payload Effects Analysis Study has identified the major cost drivers that can and must be manipulated to reduce the cost of payload operations; these are: (1) relaxation of weight and volume constraints; (2) acceptance of higher risk made possible by the advantages of Space Shuttle: payload retrieval,

refurbishment and reuse, and (3) spacecraft standardization. The study also provided an initial quantification of the cost savings. It was found that these effects are additive, and that standardization as an additional Payload Effect, can have a strong effect on the reduction of RDT&E costs. The overall results are so encouraging that they have not just served as inputs for Mathematica's analysis and endorsement of the Space Shuttle/Space Tug project, but also promise that for given expenditures, payload activities can be considerably extended.

It has been shown that spacecraft standardization can materially cut peak RDT&E funding requirements. This supplements the savings which result from program-peculiar low-cost payload designs.

With these large saving potentials, the question of subsequent activities becomes: (a) How soon can the low-cost payload techniques, including standardization, be reduced to hardware?, and (b) what are the most logical steps to take?

The objectives to which future activities should be directed are:

(1) Strengthen the data base of unmanned spacecraft by:

- including more scientific, and commercial, near-earth and planetary satellites

- improvement of the cost estimating relationships and cost projection methods

(2) Widen the scope of Payload Effects Analysis to include:

- manned system payloads
- lunar surface exploration payloads

- propulsive stages, either in their capacity as payloads, or transportation systems
- Shuttle-carried payload equipment for on-board checkout

(3) Implement spacecraft standardization by:

- selection and optimization of type and performance variance of subsystem options
- conceptual design and evaluation of various standardized spacecraft approaches
- conceptual design and evaluation of subsystems and interface hardware for selected standardized spacecraft concept(s).

In view of the relatively short time span to the need date of Shuttle IOC, it appears desirable to initiate these follow-on activities in the very near future.

10.3.1 Expansion of Low-Cost Payload Data Base

The low-cost data base resulting from the present Payload Effects Analysis was limited by necessity to conceptual designs of low-cost versions of the OAO-B, the SEO, and the SRS; and in cost estimates and plans for implementation. It has been recognized that not all future space payloads derive similar benefits from the application of low-cost approaches. In order to make precise forecasts of the expected space program-wide savings, an extension of this data base would be desirable. Following are some specific areas in which follow-on analysis effort is recommended to provide additional, increased-depth data in the next orderly step toward actual implementation of low-cost hardware programs.

(a) Payload Effects Analysis of Additional Unmanned Payloads

Develop low-cost designs, prepare outline development plans, and estimate comparative costs for additional payloads representing important, and different,

missions of the Traffic Model: (a) Communications Satellite, (b) Mars Orbiter or Viking, (c) OSO, (d) Large Space Telescope, and ERTS.

(b) Logistics and Cost Analysis of Payload Maintenance and Refurbishment

Develop rationale and procedures for establishing minimum-cost payload programs based upon the optimum combination of spare equipment modules, maintenance visits, complete refurbishment of payload, and ground refurbishment and reuse of failed/used modules. Quantify cost data and establish methodology for applying the data to the total Fleet Analysis.

(c) Additional Analysis of Reduced Reliability and Costs

Quantify, in terms of payload reliability and confidence level, and in payload program costs; the benefits offered by the Shuttle in (a) reducing or eliminating the payload launch/ascent failures and (b) reducing the payload reliability (and cost) as a result of access to the payload for on-orbit repair.

(d) Cost Effects Analysis of Simplified NASA Program Requirements

Establish the degree of influence of NASA Program requirements upon typical unmanned payload program costs and identify areas in which requirements could be altered by use of the Space Shuttle, such as documentation, design and testing specifications, hardware traceability, and reliability demonstration tests.

(e) Cost Optimization Methodology and Application

During the initial Payload Effects Study, a special methodology was developed by which payload programs could be cost-optimized, taking advantage of relaxed weight/volume constraints, lower launch costs, on-orbit repair/refurbishment, and retrieval and reuse. Because of new concepts and economic data, this methodology can now be expanded and improved. Incorporate the findings of the above studies into the payload cost optimization program. Add summation capability to computer programs so groups of missions in total Traffic Model can be traded off.

10.3.2 Increase of Scope of Payload Effects Study

The initial Payload Effects Study was concerned exclusively with unmanned payloads. There are a number of circumstances for which Payload Effects may be distinctly different from the ones obtained thus far.

(a) Payload Effects Analysis - Earth and Lunar Orbit Manned-System Payloads

Manned payloads, with their larger volumes and weights, are more likely to be constrained by Space Shuttle/Space Tug geometry and performance than unmanned payloads. Yet, within the geometrical constraints, very attractive modular designs are possible. This and sharply reduced transportation costs are expected to result in considerable savings. In order to assess the related Payload Effects, analyze a selected group of payloads, determine the applicability of low-cost design and program approaches, determine the payload/Shuttle/Tug interfaces, and estimate the payload cost reduction possible. The payloads will include: Space Station segments, Personnel/Cargo Modules, Space Tug Manned Module, Large Payloads with Crew Stations, Space Rescue Modules.

(b) Payload Effects Analysis - Lunar Surface Exploration Payloads

For high earth orbits or for lunar missions there will be quite a large decrease in transportation cost in going from the historical (expendable-launched) payloads to the Shuttle-launched. This will provide a strong reduction in transportation/payload cost ratio and implies that reoptimization of payload is indicated. In view of the fact that transportation costs to high orbits and to lunar orbit or the lunar surface vary proportionally with earth-to-low orbit transportation costs, and that the design of payloads in this category has been forced to considerable sophistication; it may be expected that the cost savings resulting from Payload Effects could be greater (in absolute dollars) than for low earth orbital missions.

In order to evaluate the related payload effects, investigate:

- Various types of transportation vehicles, including lunar landing vehicles operating between earth orbit, lunar orbit and lunar surface; lunar flyers and lunar surface rovers; and lunar rescue vehicles.

- Various lunar base equipment, for both manned and unmanned exploration, including: lunar shelters, power plants, scientific experiments and equipment, and related surface logistics items.

- Perform a parametric cost evaluation of alternative ways to implement the planned lunar surface payload programs, identify the most cost-effective approaches, and estimate cost savings (compared with planned mission costs). All transportation to be based on or be compatible with the Space Shuttle.

(c) Induced Payload Effects of Special-Purpose Unmanned Payloads

In addition to those payload effects that result from the conversion of historical or existing designs for use on a low-cost space transportation system, there are induced payload effects resulting from the use of the Space Shuttle:

- The ratio of benefit-to-investment of planned payloads becomes attractively larger, resulting in savings, which in turn can be applied to additional quantities of similar payloads which were excluded initially because of budget limits and their lower benefit ratio.

- The Shuttle will accommodate a group of payloads which are incompatible with expendable launch systems.

The second induced effect will allow sufficiently different payloads to be carried to drastically change the operational scope of space operations.

In order to evaluate these payload effects, conduct a general Payload Effects analysis for special payloads such as propulsion stages, propellant tankage, vehicle segments for orbit assembly, multiple payloads. Determine special payload/Shuttle interfaces and estimate the potential cost savings resulting from application of low-cost approaches.

(d) Preliminary Design and Costing of Shuttle-Carried Payload Checkout Equipment

During the initial Payload Effects Study, the concept of checking out payloads while mounted in the Shuttle and using a Shuttle-carried payload checkout set was developed. Preliminary assessments of feasibility were made and preliminary checkout lists tailored for on-board Shuttle checkout, were prepared for the OAO-B, SEO, OSO, and OGO. Savings made possible by orbital checkout are included in the "Risk Acceptance" payload effect.

Preliminary volumes and weights were determined for a Shuttle-mounted checkout set and very rough cost estimates made for the hardware and software.

Because of the critical importance of this checkout interface, which has a strong influence on payload design as well as on establishment of operational concepts with the Shuttle and payloads; it is necessary that additional study be accomplished to validate the operational feasibility and economic considerations of using a Shuttle-carried payload checkout set.

In order to evaluate the related payload effects, develop the concept of a payload checkout set which has multi-program usage and verify its practicality and compatibility with the Space Shuttle operation. Estimate costs on the hardware, software, and operations.

(e) Preliminary Design and Costing of Universal Payload Deployment/Retrieval Installation for Shuttle

Develop a conceptual design for and estimate costs of a universal-usage gear which can be used by all or most unmanned payloads carried by the Shuttle. Apply low-cost design techniques.

10.3.3 Implementation of Spacecraft Standardization

It was demonstrated that the introduction of standardized subsystems, with application to a number of different payloads, can be extremely effective in reducing overall payload development costs.

The concept of standardized subsystems has many benefits; most are directly convertible to cost reductions: single development cost, production in relatively large quantities, simplified test requirements, reduction in payload development time and risk, over-sized subsystems to allow for payload performance growth, easy replaceability, reduced cost of field repair and refurbishment.

On a higher level of standardization, it appears economically desirable to implement a series of standard spacecraft, each of which could support a fairly wide spectrum of planned missions. Standard subsystem modules could be used as "building blocks" in constructing a standard spacecraft.

In order to make modular design with many possible design options, interface requirements from subsystem to subsystem and also from payload experiments to spacecraft have to be established carefully in order to avoid interference and to assure future growth potential. This results in somewhat increased development costs. However, it appears that this effect can be minimized by selecting an optimum variance of equipment options. Based on the present study, it appears that both specialized as well as multipurpose spacecraft could be assembled from the same standardized modular subsystems options.

In order to realize standardized spacecraft design at an early time, it is important to perform the following studies in the very near time frame:

(a) Selection of Type and Variance of Standardized Subsystems Options

Survey mission model to establish incidence and spread of subsystems performance requirements. Select modular equipment options to satisfy requirements. Perform optimization of type and variance on a total program cost basis.

(b) Conceptual Design and Evaluation of Various Standardized Spacecraft Approaches

Based on standardized modular subsystem options, configure various spacecraft approaches (in the following categories) to satisfy the requirements of various segments of the mission model:

(1) Special-purpose spacecraft, configured to a single set of payload requirements, but assembled from standardized subsystems.

(2) Multipurpose spacecraft, configured to cover the potential requirements of several, alternate payloads.

(3) Cluster spacecraft, configured to cover the potential requirements of a spectrum of simultaneous payloads.

With an increasing number of payloads to be accommodated, spacecraft overdesign (relative to any single mission) increases. However, in view of payload refurbishment and reuse, and when considering the effect of social discounting, it is not clear which approach or combination of approaches is economically the most useful. Thus a Phase A type study of standardization is needed.

(c) Conceptual Design and Evaluation of Subsystems and Interface Hardware for Selected Standardized Spacecraft Concept(s)

Based on the results of the evaluation of standardized spacecraft approaches, the physical design of the selected subsystems modules will be carried out to conform with the various subsystem/subsystem and subsystem/payload as well as man/machine interface requirements, and to ensure true modular growth capability.